Delta Launch Vehicle
Secondary Payload Planner's Guide
For NASA Missions

September 1999

<u>Please note</u>:* *this document is currently under revision and some sections may be missing data or slightly out of date, but overall, it is still a good general reference for Secondary payload accommodations on the Delta II LV. Please call NASA/KSC/ELV Launch Services with any questions/clarifications.

National Aeronautics and Space Administration
Kennedy Space Center
ELV Launch Services Directorate

This document supersedes the Secondary Payload Planner's Guide dated November 1993

Preface

A Delta secondary payload is a payload sufficiently small in volume and mass that it can be carried in space available between the Delta second stage and the fairing without impact to a primary mission. This Delta Secondary Payload Planner's Guide provides procedural and technical information to payload projects proposing secondary payloads as NASA or NASA-sponsored missions.

Inquiries relative to this Guide should be directed to the
NASA/KSC ELV Launch Services
Mail Code: VB-B2
KSC, FL 32899
407-476-3622 or 407-853-5761
Fax: 407-853-4357

The Overview section of the Guide describes the process for acquiring a flight as a secondary payload on Delta and the capabilities and restrictions for flight. Also, the Overview notes those areas of the Guide which are of particular relevance to secondary payloads. The Guide provides details of interface requirements and integration procedures.

Contents

Overview

The Delta Secondary Payload Planner's Guide is published by the NASA/KSC/ELV Launch Services, to provide interface information for secondary payload (SP) projects. Because it is intended to be a single-source reference, pertinent sections of the Delta II Commercial Payload Planners Guide (Reference 1) are included.

Procedures for Obtaining Approval to Launch

Under a Memorandum of Agreement between NASA and the USAF, civil Government SPs may be launched on USAF missions, and NASA KSC Mission Integration Manager will be the program interface for these payloads.

Projects proposing to launch SPs on NASA or USAF payload missions should follow these procedures:

- Contact the NASA/KSC/ELV Launch Services to determine availability of a Delta launch of acceptable orbit parameters, of adequate mass margin, and to be launched in a suitable time frame. Provide the following information.

    - Brief description of the payload mission
    - Brief description of the science or application benefits to the U.S. Government from the proposal
    - Mass and dimensions of payload
    - Requirement to separate from or stay with the Delta stage
    - For non-separating payloads, minimum acceptable duration of the mission
    - For non-separating payloads, support required from the vehicle during the mission (eg. TM, battery power, sequence commands)
    - Payload constraints (desired orbit, launch window, sun angles, mounting orientation, etc.)

The information should be provided to NASA/KSC/ELV:

NASA KSC
ELV Launch Services
Mail Code: VB-B2
KSC, FL 32899
Attn: Darrell R. Foster
Phone: (407) 476-3622, or (407) 853-5761
Fax: (407) 853-4357
Email: darrell.foster-1@ksc.nasa.gov

- After establishing availability of a compatible launch opportunity, submit the application form at the end of this section to

  NASA Headquarters
  Expendable Launch Vehicle Requirements Office
  New Programs and Integration / Code SV
  Washington, D.C. 20546
  Fax: (202) 358-4163

The Office of Expendable Launch Vehicles will process the application and seek a sponsor organization for the secondary payload from within NASA. If a sponsor for the payload is obtained, NASA will fund all or part of the cost of integration of the payload with the vehicle. To expedite the SP project may want to seek NASA sponsorship prior / in parallel with contacting KSC and HQ.

Overview of Secondary Payload Services and Constraints

Prior to application, the prospective SP project should review the following services and constraints for compatibility with the proposed mission:

- Constraints
  - The SP shall present no hazard, such as ordnance, radiation, or contamination, to the primary payload.
  - Acceptance of the SP is subject to the approval of the primary payload project manager.
  - Primary mission orbit requirements shall not be affected by SPs.
  - Approval of SPs will be considered only if sufficient performance margin exists for the primary mission and could be withdrawn if the margin is unexpectedly reduced.
  - The SP shall not impact the primary mission's ability to launch on time.
  - The SP may not intrude on the primary payload clearance envelope. Existing mounting hardware designs accommodate a variety of sizes and shapes of SPs. <u>However, in general, one dimension of the SP must be less than 31 cm (12 in) for separable SPs or 38 cm (15 in) for non-separating SPs. Envelopes are about 0.5 in larger for NASA primary payload missions.</u>

- SPs shall incorporate dedicated power, sequencing, and wiring isolated from the primary payload. Support components may be provided by the vehicle contractor, The Boeing Company (Boeing)

- New/revised structure to accommodate SPs shall be designed to an ultimate factor of safety (f.s.) of 2.00 if not tested or 1.25 if tested; to a yield f.s. of 1.65 if not tested or 1.10 if tested.

- The support structure, clamp bands, and separation systems for SPs will be provided by Boeing or be of proven design.

- SP telemetry can be routed through the Delta II antenna and shall have separate verification.

- Support from the second stage, such as power, pointing, or separation command, is generally limited to a mission duration of about one orbit revolution. SP power on and other vehicle support is permitted after separation of the primary payload. Details on limitations on mission duration are provided in Section 2.1.

- The SP must comply with range safety requirements as described in Section 9.

- Sevices

    - Spacecraft integration support is provided by KSC and Boeing. The integration process normally begins about 2 years before the scheduled opportunity. However, longer or shorter schedules may be supportable is required. The process and typical schedule of required documentation is shown in Section 8.

    - KSC and Boeing define flight and test environments during the integration process. Typical levels are defined in Section 4.

    - A number of existing mounting hardware designs are available and are described in Section 5. Variations from these existing designs will generally increase the mission integration costs. SP Projects are encouraged to use existing designs.

Application for Launch as a Secondary Payload on Delta

Organization:
Address:

Contact:
Telephone:
Email:
Payload Name:

Estimated payload cost:

Payload Objective:

____
____
____

Support type:

- ☐ **Self-sufficient attached**
- ☐ **Attached with vehicle support**
- ☐ **Self-sufficient free-flyer**
- ☐ **Requires free-flyer bus**

Payload characteristics:

	Mass (lb / kg)	Shape	Maximum Dimensions (in/cm) (Height; Length and Width or Diam.)
Unit 1	________	______	____________________
Unit 2	________	______	____________________
Unit 3	________	______	____________________

Range of acceptable orbit parameters:

Apogee altitude (km)	________	**to**	________
Perigee altitude (km)	________	**to**	________
Inclination (deg)	________	**to**	________
Arg. Of Perigee (deg)	________	**to**	________

Attached payload required mission duration (minutes/hours):

Unique payload constraints:

____
____

Range of acceptable launch dates:

Latest access to payload necessary:

Network/telemetry requirements:

Mail to: **NASA Headquarters**
Expendable Launch Vehicle Office / Code SV
Washington, D.C. 20546
Fax to: **(202) 358-4163**

Section 1
The Delta Vehicle

1.1 DELTA LAUNCH VEHICLES

The Delta launch vehicle is currently being offered by The Boeing Company (Boeing) in two models and with two and three stage configurations:

Delta II 7320 (two-stage with 3 strap-on motors)
Delta II 7920 (two-stage with 9 strap-on motors)
Delta II 7325 (three-stage with 3 strap-on motors)
Delta II 7925 (three-stage with 9 strap-on motors)

The first digit of the four-digit system used to identify Delta configurations identifies the booster as the Delta II series, the second digit indicates the number of Hercules graphite epoxy motors (GEMS) used for strap-ons, the third digit identifies the second stage, and the fourth digit identifies a third stage.

1.2 VEHICLE CONFIGURATION

An expanded view of the vehicle is shown in Figure 1.2.

First Stage. The first stage of the 7000 series vehicle configuration uses the Rocketdyne RS-27 engine with a 12:1 expansion ratio and the Hercules GEM solid rocket strap-ons. The RS-27 is a single-start, liquid bipropellant rocket engine with a thrust rating of 207,000 lb (921 Kn) at sea level. Two vernier engines provide roll control during main-engine burn, and attitude control after cutoff and before second-stage separation. Thrust augmentation is provided by three or nine solid propellant rocket motors. In the nine-solid case, six are ignited at liftoff and the remaining three are ignited in flight.

Interstage. The Delta interstage assembly extends from the top of the first stage to the second-stage miniskirt. It carries loads from the second stage, third stage, spacecraft and fairing to the first stage, and contains an exhaust vent and six spring-driven separation rods.

Second Stage. The second stage uses the restartable Aerojet AJ10-118K engine with nitrogen tetroxide and Aerozine-50 storable propellants. Gaseous helium is used for pressurization, and a nitrogen cold gas jet system provides attitude control during coast periods and roll control during powered flight. Hydraulically-activated actuators provide pitch and yaw control during powered flight. The forward section of the second stage houses guidance and control equipment that provides guidance sequencing and stability control signals for both first and second stages. The Delta inertial guidance system (DIGS) is a strap-down all inertial system consisting of a Delta redundant inertial measurement system (DRIMS) and a guidance computer (GC). DRIMS data is processed in the computer to obtain attitude reference and navigation information. The computer issues preprogrammed sequence commands and provides control system stabilization logic for both powered and coast phases of flight. The vehicle also contains a telemetry system and a range-safety tracking system.

Third Stage. For missions requiring three stages, the third stage is the Star-48B solid-rocket motor, which is supported at the base on a spin table that mates to the top of the second-stage guidance section. The payload attach fitting (PAF) is the structure that provides the transition from the top of the solid-rocket motor to the spacecraft interface. Before third-stage deployment, the stage and spacecraft are spun-up using spin rockets rotating the assembly on a spin bearing. An ordnance sequencing system is used to release the third-stage/spacecraft after spin-up, to ignite the Star-48B motor, and to separate the spacecraft following motor burn. The stage also contains an S-band T/M system and a nutation control system (NCS) to suppress coning.

Payload Fairing. The final vehicle element is the fairing, which shields the payload from buffeting and aerodynamic heating while in the atmospheric phase flight. Two fairings are typically flown, identified by overall diameter: 9.5-ft and 10-ft. Fairings are discussed in detail in Section 3.

1.3 VEHICLE AXES/ATTITUDE DEFINITIONS

The vehicle centerline is the longitudinal axis of the vehicle. Axes I, II, III, and IV are defined as shown in Figure 1.3, with axis II being the downrange side of the vehicle and axis IV the up range side.

The vehicle pitch plane is defined by the vehicle centerline and axes II and IV (i.e., the vehicle pitches about the I and III axes). Positive pitch rotates the nose of the vehicle up, towards the IV axis.

The vehicle yaw plane is defined by the vehicle centerline and axes I and III (i.e., the vehicle yaws about the II and IV axes). Positive yaw rotates the nose of the vehicle to the right, towards the I axis.

The vehicle rolls about the centerline, with positive roll being rotation in a clockwise direction, looking forward from the aft end of the vehicle, (i.e., from axis I towards II). The third-stage spintable also spins in the same direction (i.e., the positive roll direction).

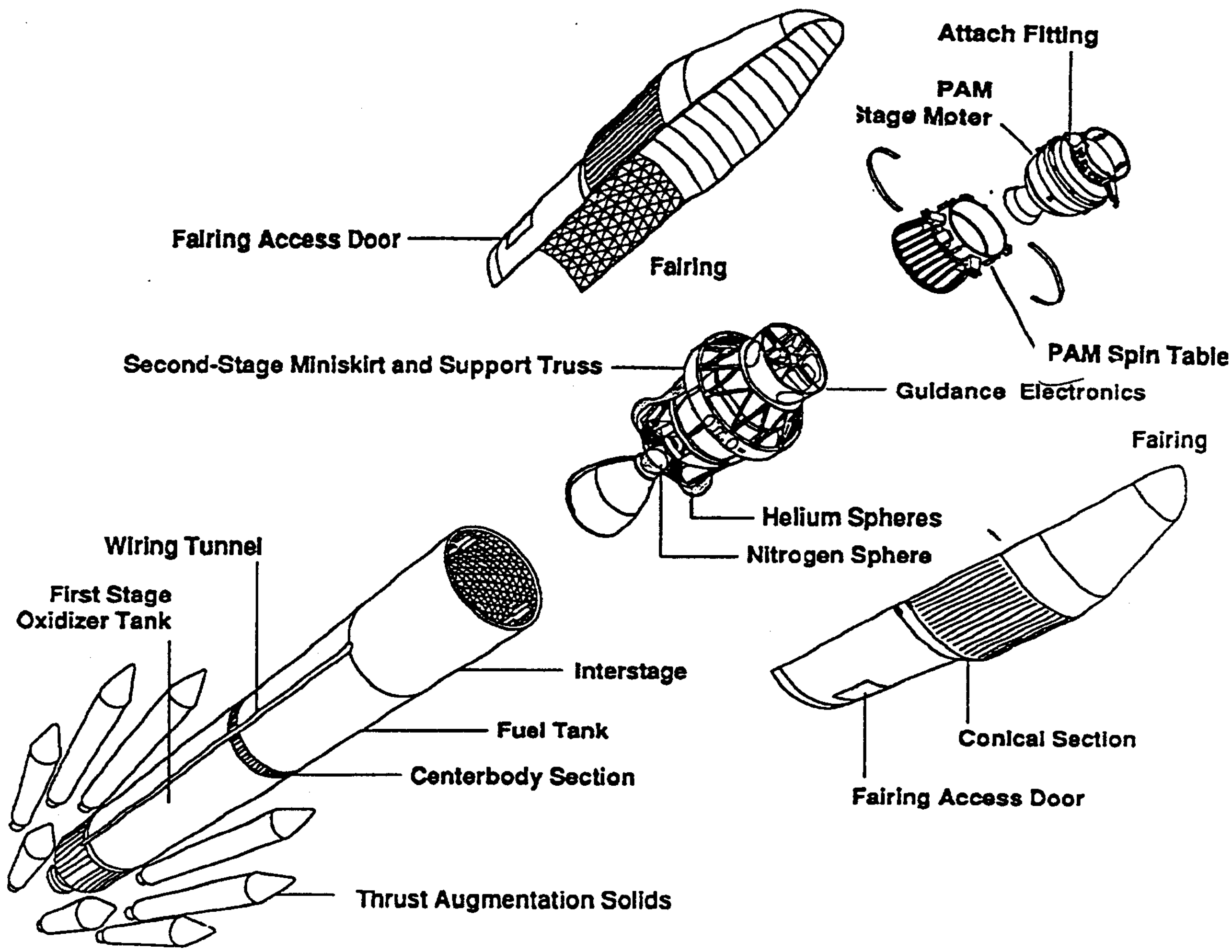


Figure 1.2 Delta II 7925 Vehicle

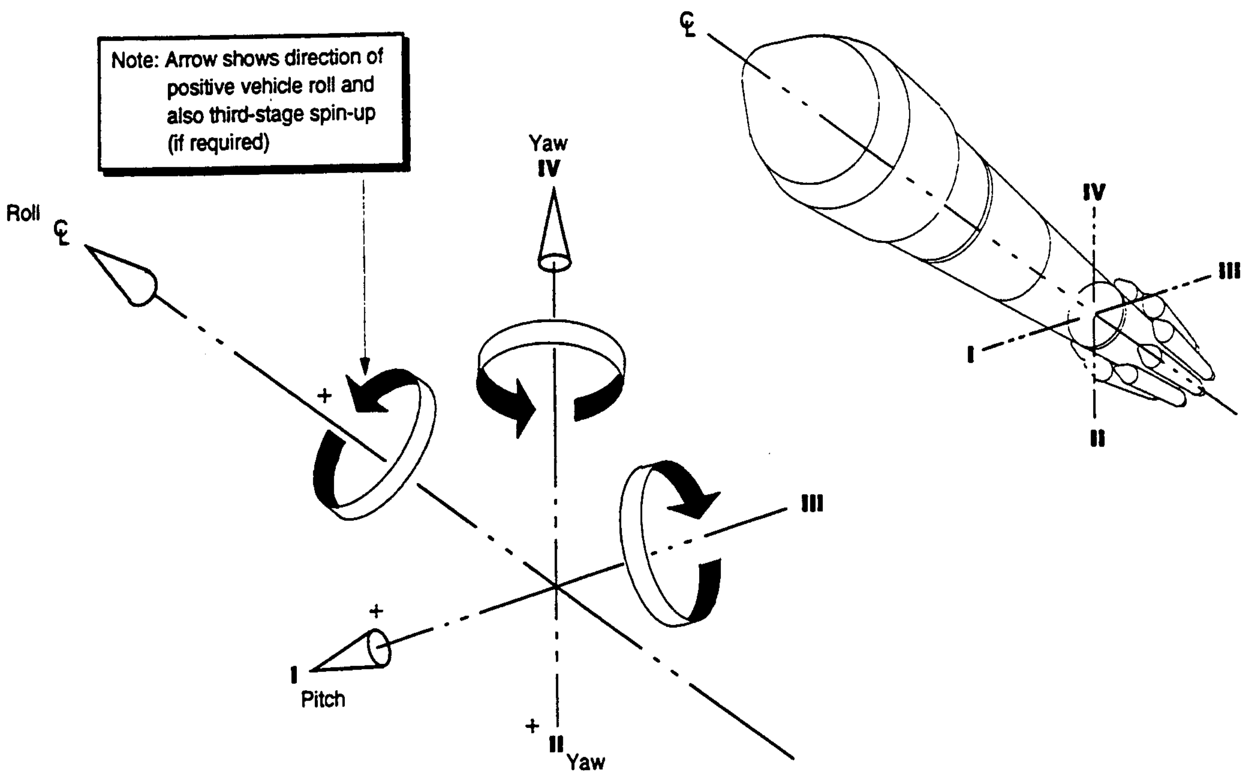


Figure 1.3 Vehicle Axes

Section 2
Performance

2.1 MISSION PROFILES

Mission profiles for both the two-stage and three-stage missions are shown in Figure 2.1. In both cases, for the nine-solid configuration, the first stage RS-27 engine and six of the solid-rocket motor boosters are ignited on the ground at liftoff. Following burnout of the six solids, the remaining three are ignited, and the six spent cases are then jettisoned about one second later, three at a time. The remaining three solids are then jettisoned about one minute later. The main engine then continues to burn until main engine cutoff (MECO).

Following a short coast period, the first- to second-stage separation bolts are fractured, followed by second-stage ignition approximately five seconds later. The next major event is the fairing separation, which occurs early in the second-stage flight.

In the typical two-stage circular orbit mission (Figure 2.1), the second stage burns for approximately 350-380 seconds, at which time stage two engine cutoff (SECO 1) occurs. The vehicle then follows a Hohmann transfer trajectory to the desired low Earth orbit altitude. Approximately 2900 seconds after SECO 1, the second stage is re-ignited and completes its burn placing the payload in the desired circular orbit. Following SECO 2, the second stage is reoriented to the desired separation attitude. Spacecraft separation is approximately 250 seconds after SECO 2.

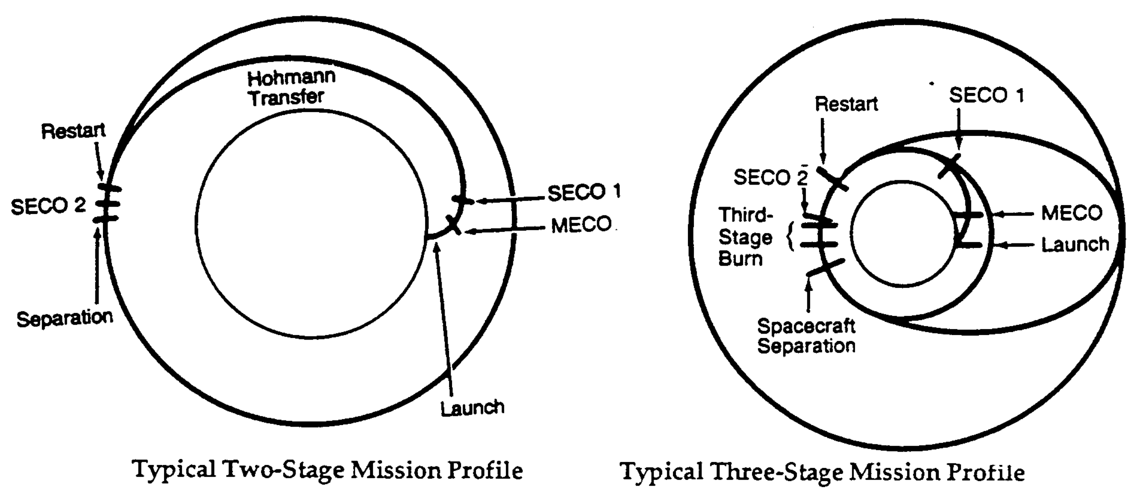


Figure 2.1 Mission Profiles

The second stage then performs an evasive maneuver, which is a short burn to increase distance from the spacecraft. Typically at least 15 minutes later, a final burn depletes propellants to avoid their possible long-term interaction, which could result in a spontaneous explosion of the stage and orbital debris.

A free-flying secondary payload is typically separated from the second stage after completion of the primary mission, the evasive burn and before the depletion burn, which must be performed prior to 7,000 seconds with the standard thermal control system. An attached secondary payload requiring second stage support (pointing, power, telemetry, etc.) is limited in mission duration by the life of the stage batteries and control gas. Total second stage operational capability is limited to about 9,000 seconds from liftoff. Vehicle telemetry, but not control, can be extended to about 30,000 seconds as a maximum with an easy modification. The second stage evasive and depletion burns may be employed to transfer the secondary payload into its desired orbit. The extent of the orbit change achievable via these burns is dependent on the primary payload performance margin. Orbit errors resulting in the use of these burns are typically greater than errors in a primary payload orbit. If control gas margin is available, the second stage could perform a roll-up maneuver to provide attitude stability for a limited time after the depletion burn.

The typical three-stage mission (Figure 2.1), uses the first burn of the second stage to place the payload into a 100 nmi (185 km) parking orbit at SECO 1. The vehicle then coasts to a pre-determined position, at which time the second stage restart burn occurs. The third stage is then spun-up, the second stage and spintable separated from the stack, and the third stage burned to establish the transfer or final orbit. Depending on mission requirements and payload mass, some inclination can be removed via the burn sequence out of the Earth parking orbit. The second stage orbit resulting from the restart burn is typically a fairly low eccentric orbit, with apogee depending on the length of the burn required for the primary mission. A depletion burn is performed after separation of the primary payload.

In summary, separating SPs will be separated after completion of the primary mission and prior to 7000 seconds. Attached SP operation will generally be limited to the period between completion of the primary mission and 9000 unless the payload can function beyond these limits with limited vehicle support (telemetry only).

2.2 PERFORMANCE CAPABILITY

The NASA/KSC/ELV Project maintains a database of primary mission performance requirements and margins. Availability of launch opportunities with sufficient margin are determined from this database and provided to payload projects and NASA Headquarters in support of secondary payload applications for launch.

2.3 ORBIT ACCURACY

The Delta Inertial Guidance Systems (DIGS) is mounted in the second-stage guidance compartment. This system provides precise attitude pointing and injection altitude accuracy for both two- and three-stage missions.

The typical two-stage mission is flown to a low-altitude circular orbit using a Hohmann-transfer mode between second-stage burns, i.e., the second stage is restarted after a coast period to circularize the orbit. In these cases the accuracy achieved by the DIGS is precise. Typical three-sigma dispersion values are less than 5 nmi (9 km) in apogee or perigee, about 2 nmi (4 km) in semi-major axis, and less than $\pm$ 0.04 deg in orbit inclination for a nominally circular orbit.

On a three-stage mission, although the parking-orbit parameters achieved are quite accurate, the final orbit is primarily affected by the third stage pointing error and the impulse error from the spin-stabilized third-stage solid motor burn.

In either case, the depletion burn of the second stage produces a sizable dispersion in orbit elements because of the uncertainty as to the amount of reserve propellants remaining after elimination of normal vehicle variations during powered flight. For missions without SPs, the depletion burn is designed to preclude re-entry of the stage, and velocity is expended primarily in corss-track with a change in inclination of one or two degrees. A typical 3-sigma uncertainty for the second stage orbit after the depletion burn is highly dependent on the launch vehicle performance margin including the SP(s) mass(es). Performing the depletion burn before the SP operation may provide additional time for second stage support of the SP operation.

Section 3
Fairings

3.1 THE 9.5-FT (2.9 M) DIAMETER SPACECRAFT FAIRING

The spacecraft is protected by a fairing which shields it from buffeting and aerodynamic heating while in the lower atmosphere. The fairing is jettisoned during second-stage powered flight at an altitude of at least 67 nmi (125 km).

The 114-in. (2896 mm) Delta fairing (Figure 3.1) is 334.2-in. (8488 mm) long and varies in diameter from 96 in. (2438 mm) at the base to 114 in. (2896 mm) at the widest portion. The fairing is an all-aluminum structure fabricated in two half-shells. These shells consist of a hemispherical nose cap with a biconic section tapering from the nose cap to a 114-in. (2896 mm) diameter. The biconic section is a ring-stiffened monocoque structure; one-half of which is fiberglass with a removable aluminum foil lining. The cylindrical base section is an integrally stiffened isogrid structure at the 96-in. (2438 mm) diameter section while the 114-in. (2896 mm) diameter cylindrical center section is skin and stringer construction.

Acoustic absorption blankets are provided within the faring interior from the nose cap to approximately Station 491. These blankets vary in thickness: 1.5 in. (38.1 mm) in the nose section, 3.0 in. (76.2 mm) in the 114-in. (2896 mm) diameter section, and 1.5 in. (38.1 mm) in the upper portion of the 96 in. (2438 mm) diameter section. The acoustic blankets consist of a 1.5 in. (38.1 mm) to 3.0 in. (76.2 mm) thick silicone-bonded heat-treated glass-fiber batt enclosed between two 0.003 in. (0.076 mm) conductive teflon-impregnated fiberglass face sheets and are heat-seal bonded. The blankets are vented through a 5 micron stainless steel mesh filter, which controls particulate contamination. Outgassing of the acoustic blankets meets the criteria of 1.0% maximum total weight loss and 0.10% maximum volitile condensable materials. Acoustic blankets are not installed in the SP region. Environments and envelopes for SPs are given for this condition.

The half-shells are joined by a contamination-free linear piston/cylinder thrusting separation system that runs longitudinally the full length of the fairing. Functionally redundant explosive bolt assemblies provide the structural continuity at the base ring of the fairing.

The fairing half-shells are jettisoned by actuation of the base and transition separation bolts, and by the detonating fuse in the thrusting joint cylinder rail cavity. Gas pressure generated by the detonating-fuse shears the attach rivets and thrusts the half-shells laterally to attain immediate spacecraft and vehicle clearance. A bellows assembly, within each cylinder rail, retains the detonating-fuse gases to prevent any contamination of the spacecraft during the fairing separation event.

Contamination of the spacecraft is minimized by frequent cleaning of the fairing surfaces during various stages of the manufacturing process. The surfaces are vacuumed and brushed until all particles visible to the naked eye are removed. All faying surfaces in the cylindrical and conical sections are sealed to entrap any remaining contamination lodged between them. Capped nut plates are used on all removable fasteners to prevent contamination of the spacecraft during the fairing installation tasks.

An air-conditioning inlet umbilical door on the fairing provides a controlled environment to the spacecraft and launch vehicle second stage while on the launch stand. A maximum air flow of 1500 ft^3 (42.5 m^3) / min meets spacecraft specific temperature requirements within the range of 60 to 80°F (15.5 to 32.2°C) while holding the relative humidity to a maximum of 50%. The air conditioning capability is identical on all three fairing designs.

3.2 THE 10-FT (3.0 M) DIAMETER SPACECRAFT FAIRING

The 10-ft (3.0 m) diameter fairing (Figure 3.2) is available for primary spacecraft requiring a larger spacecraft envelope. This fairing (Figure 3.2) is 312.0 in. (7924.8 mm) long and varies in diameter from 96 in. (2438.3 mm) at the base to 120.3 in. (3055.6 mm) at the widest portion. The fairing separates into trisectors. It is an all-aluminum structure with the exception of a fiberglass RF window in one of the trisectors.

The conic section is a ring-stiffened semi-monocoque structure, of which one trisector is fiberglass with a removable aluminum foil lining. The 120.3 in (3055.6 mm) diameter cylindrical section is of skin and stringer construction, while the 96.0 in. (2438.4 mm) diameter cylindrical section at the base is an integrally stiffened isogrid structure. The base section conical adapter is a ring-stiffened semi-monocoque structure with an outer aerodynamic skirt extension of skin and stringer construction.

Acoustic absorption blankets are provided within the fairing interior from Station 337 to approximately Station 481. These blankets are 1.5 in. (38.1 mm) thick and are constructed of silicone-bonded heat-treated glass-fiber batt enclosed between two 0.003 in. (0.076 mm) conductive teflon-impregnated fiberglass face sheets. The blankets are vented through a 5 micron stainless steel mesh filter, which controls particulate contamination. Outgassing of the acoustic blankets meets the criteria of 1.0% maximum total weight loss and a 0.10% maximum volatile condensable material.

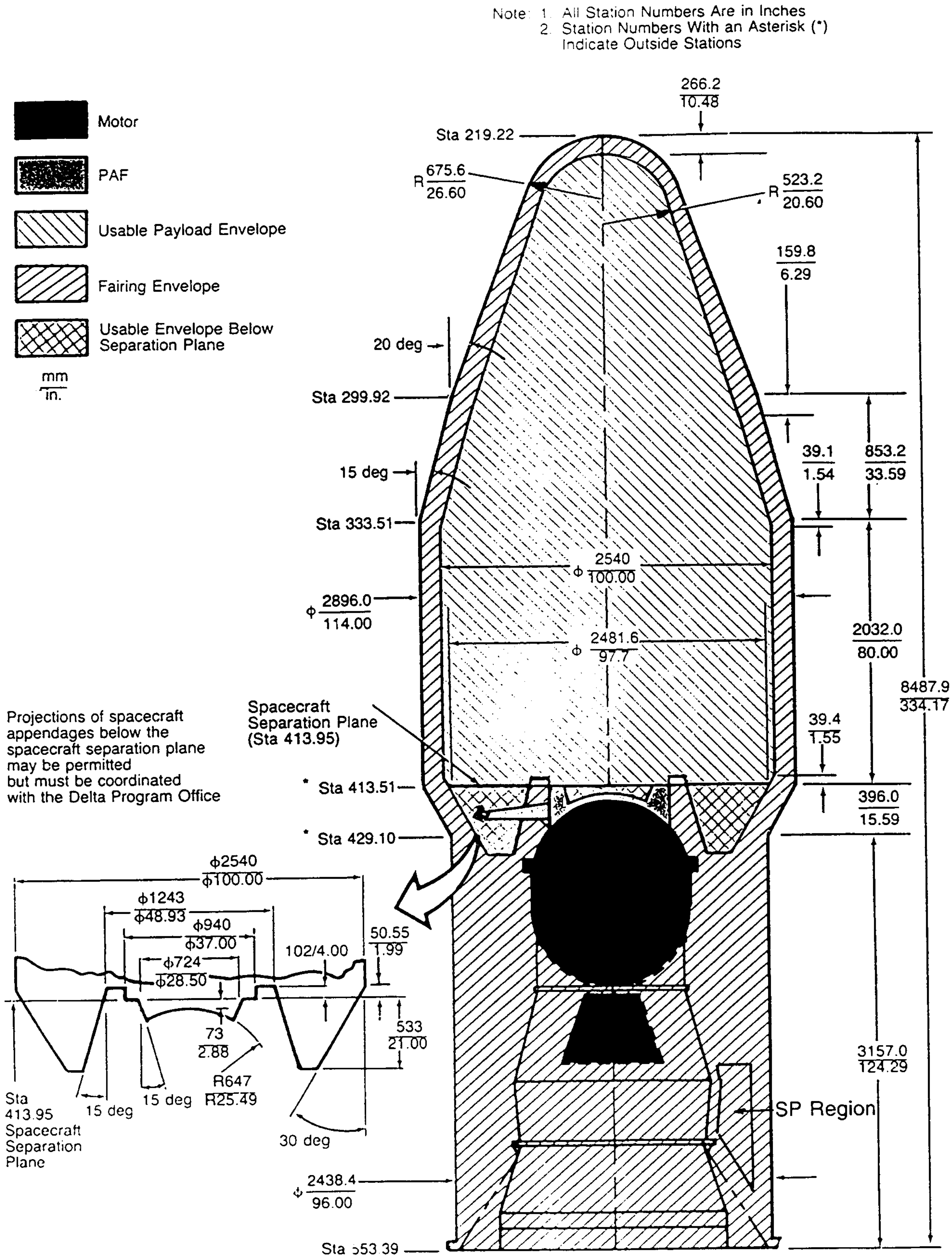


Figure 3.1a Delta 9.5-ft (2.9 m) Fairing, Three Stage Envelope

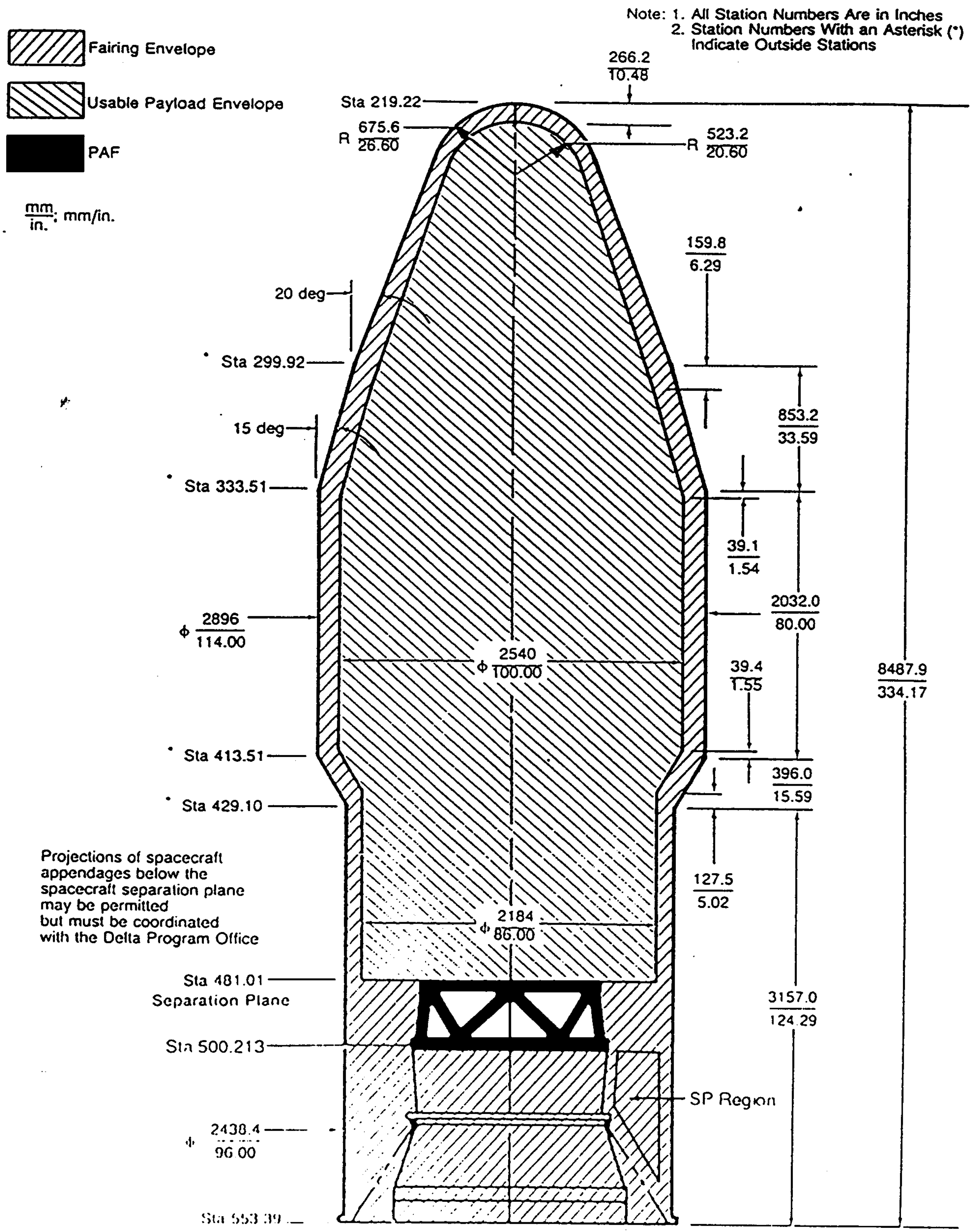


Figure 3.1b Delta 9.5-ft (2.9 m) Fairing, Two Stage Envelope (6019 PAF)

The trisectors are joined by a contamination-free linear piston / cylinder thrusting separation system that runs longitudinally the full length of the fairing. Functionally redundant explosive bolt assemblies provide the structural continuity at the base of the fairing.

The fairing trisectors are jettisoned by actuation of the separation nuts and by the detonating fuse in the thrusting joint cylinder rail cavity. Gas pressure generated by the detonating fuse shears the attach rivets and thrusts the trisectors laterally to attain immediate spacecraft and vehicle clearance. A bellows assembly, within each cylinder rail, retains the detonating-fuse gases to prevent any contamination of the spacecraft during fairing separation.

Contamination of the spacecraft is minimized by frequent cleaning of the fairing surfaces during various stages of the manufacturing process. The surfaces are vacuumed and brushed until all particles visible to the naked eye are removed. All faying surfaces in the cylindrical and conical sections are sealed to entrap any remaining contamination lodged between them. Capped nut plates are used on all removable fasteners to prevent contamination of the spacecraft during the fairing installation tasks.

3.3 THE PRIMARY SPACECRAFT ENVELOPE

The allowable static spacecraft envelopes within the confines of the fairing are shown in Figures 3.1 through 3.3 for the primary spacecraft. These figures reflect envelopes for several existing attach fittings. Usable envelopes below the separation plane must be coordinated with the Delta Program Office. Local protuberances outside these envelopes are possible with coordination and approval.

SPs are typically mounted in the space between the second stage and the fairing, near the base of the fairing and above the support truss which attaches the fairing to the second stage. Secondary payload envelopes thus depend on the particular primary payload attach structure design, the fairing diameter, and the number of stages. These are defined in Section 5, where the various attach structures are described.

While the 10-foot diameter fairing provides a larger SP envelope in some areas, other areas have reduced dimensions due to primary structure at the transition joints.

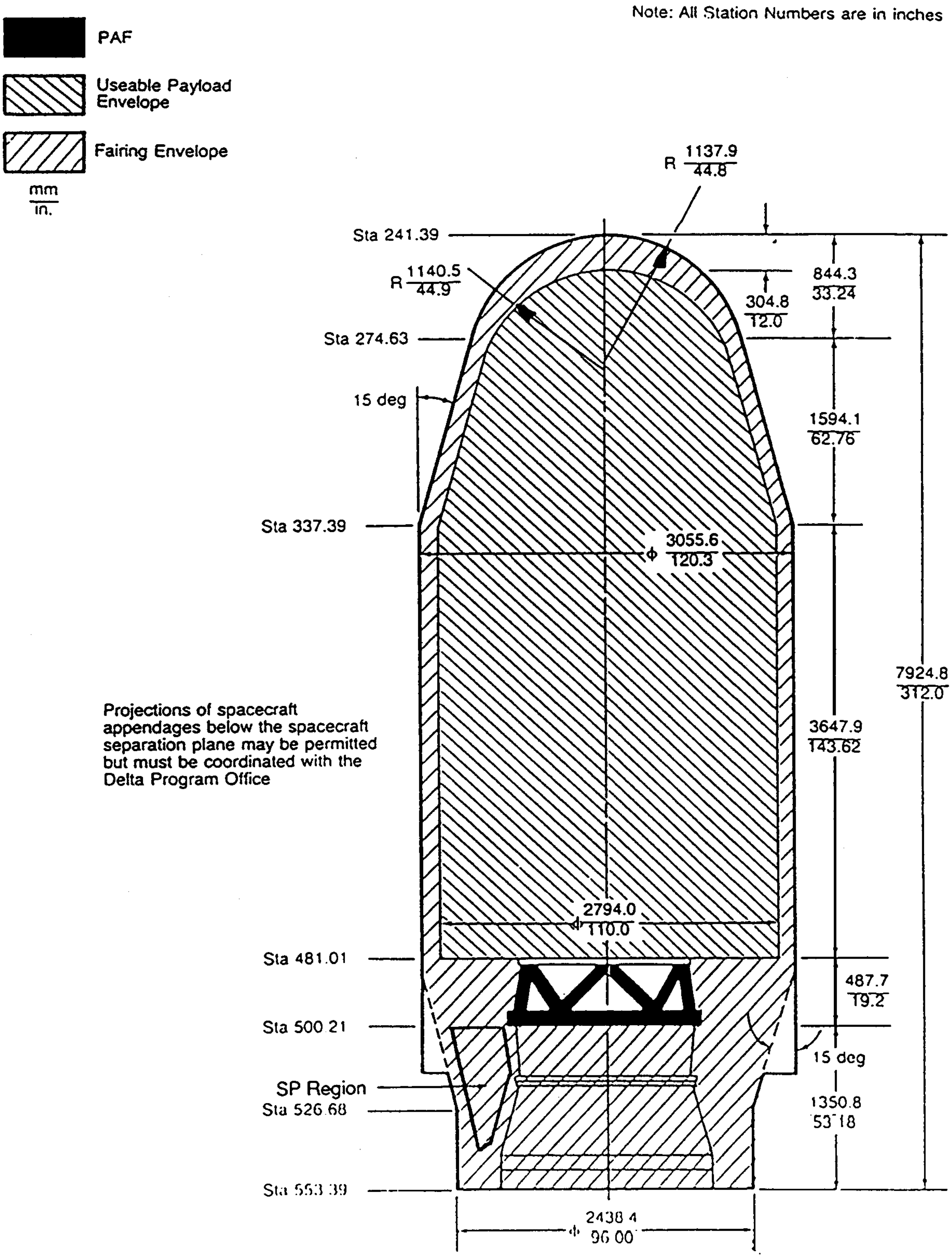


Figure 3.2a Spacecraft Envelope, 10-Ft Fairing, Two-Stage Configuration 6019 PAF

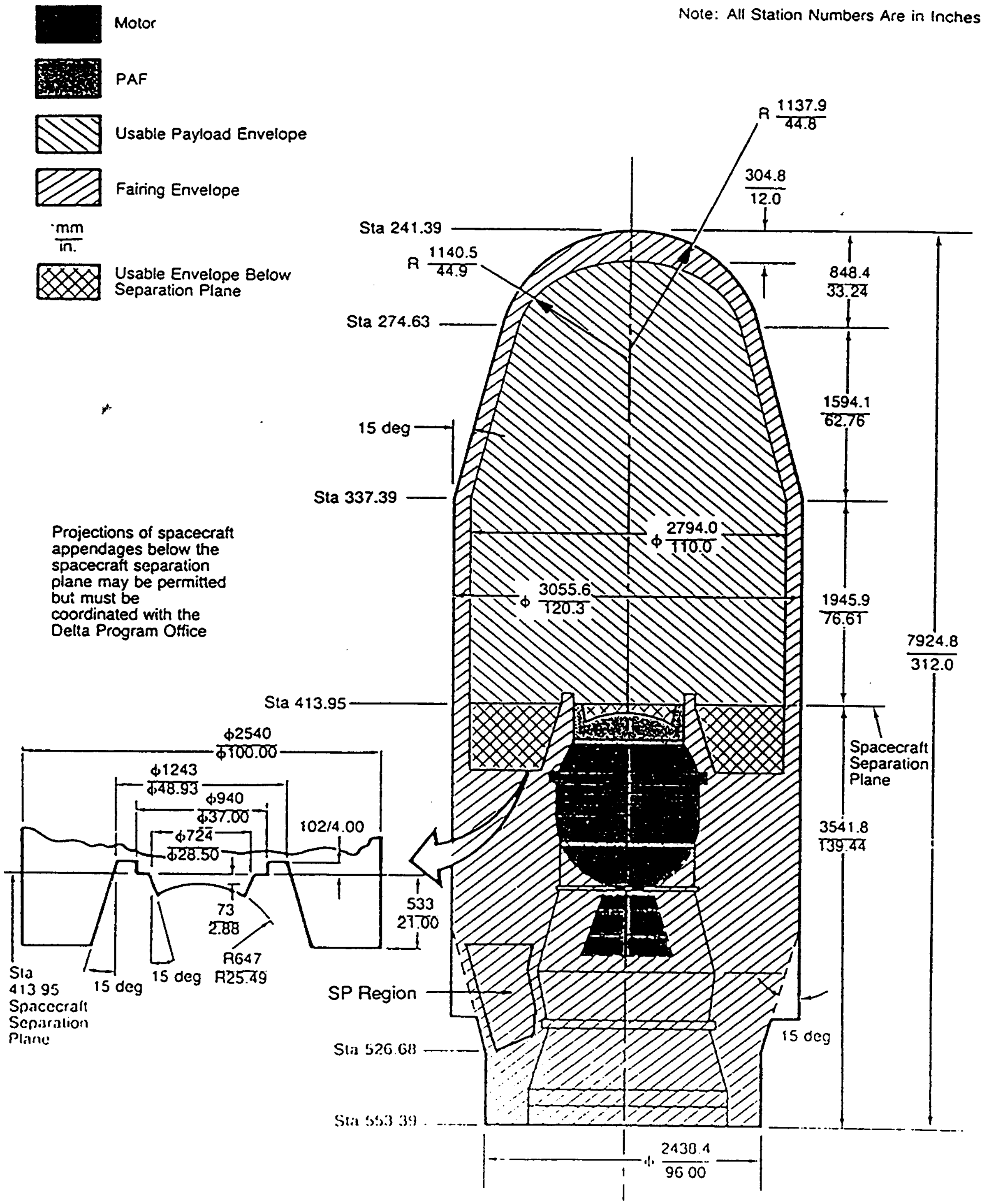


Figure 3.2b Spacecraft Envelope, 10-Ft Fairing, Two-Stage Configuration 3712 PAF

Section 4
Spacecraft Environments

4.1 PRELAUNCH ENVIRONMENTS

4.1.1 Spacecraft Air Conditioning

The environment to which the spacecraft is exposed throughout processing is carefully controlled for temperature, relative humidity, and filtration. This includes processing before mounting on the Delta II in the mobile service tower (MST) white room and the environment the spacecraft is exposed to once encapsulated inside the fairing.

Spacecraft air conditioning is supplied via an umbilical after the spacecraft and fairing are mated to the Delta II. The spacecraft air-distribution system provides air to the spacecraft at the required temperature, dew point, and flow rate. An air-conditioning inlet umbilical door on the fairing provides a controlled environment to the spacecraft and launch vehicle second stage while on the launch stand. A maximum air flow of 1500 ft^3 (42.5 m^3) / min meets spacecraft specific temperature requirements within the range of 60 to 80°F (15.5 to 32.2°C) while holding the relative humidity to a maximum of 50%. The spacecraft air-distribution system utilizes a diffuser system at the inlet of the air-conditioning duct at the fairing interface. The air-conditioning umbilical is ejected at liftoff via lanyard disconnects and the access door on the fairing automatically closes. The air supplied to the payload, at mission specific flow rates, is directed by the diffuser to flow downward and encompass the primary spacecraft, and then is vented to the atmosphere at the second stage. Flow rate, temperature, and humidity are specified by the primary spacecraft.

Environmental control specifications for the Eastern Range (ER) are listed in Table 4.1.1. Specifications for SLC-2W at VAFB are similar to those at LC17.

4.1.2 MST Tower Cleanroom

The Delta MST has provisions for clean spacecraft processing. The cleanroom in an environmentally-controlled room located at the upper levels within the MST. The cleanroom has facilities for positioning up to six separate retractable working platforms (levels) at various heights. The temperature in the cleanroom is variable from 65 to 75°F (18.2 to 23.8°C). The relative humidity is variable from 40 to 50%. The MST on launch pad 17A and the gantry at SLC-2W at VAFB are similar to 17B.

Location		Temperature	Relative humidity	Filtration
Building AE	Cleanroom complex	72 ± 3°F (22.2 ± 1.76°C)	55 ± 5%	Class 10,000*
Building AM	High bay areas Cleanroom complex (Room 117A)	75 ± 5°F (24 ± 2.8°C) 75 ± 5°F (24 ± 2.8°C)	45 ± 5% 40% max	Commercial standard ** Class 10,000
Building AO	High bay area and airlock System test areas (Room 202)	75 ± 3°F (24 ± 1.7°C) 75 ± 3°F (24 ± 1.7°C)	50 ± 5% 50 ± 5%	Class 100,000 Class 100 Horizontal Laminar flow enclosure
Handling cans	Mobile	Not controlled	Not controlled***	Not controlled
MST	LC-17A and LC-17B white Room (all doors closed) Fairing	75 ± 5°F (24 ± 2.8°C) Any specified between 60 and 80 ± 5°F (15.5 and 32.2 ± 2.8°C)	45 ± 5% 50% max ****	99.97% of all particles over 0.3µm 99.97% of all particles over 0.3µm
SAEF 2	Airlock High bay Low bays Test cells	71 ± 6°F (21.7F (21.7 ± 3.3°C) 71 ± 6°F (21.7F (21.7 ± 3.3°C) 71 ± 6°F (21.7F (21.7 ± 3.3°C) 71 ± 6°F (21.7F (21.7 ± 3.3°C)	50% max 50% max 50% max 50% max	Class 100,000 Class 100,000 Class 100,000 Class 100,000
Cargo hazardous Processing facility	Airlock Hazardous operations bay	70 ± 5°F (21 ± 2.8°C) 70 ± 5°F (21 ± 2.8°C)	30 to 50% 30 to 50%	Class 100,000 Class 100,000

*Class 1,000 obtainable with restrictions
**Class 100,000 obtainable with restrictions
***Dry gaseous nitrogen purge available if required
****50% relative humidity can be maintained at a temperature of 16.7°C (62°F). At higher temperatures, the relative humidity can be reduced by drying the conditioned air to a minimum specific humidity of 40 grains of moisture per 0.45 kg (1 lb) of dry air.

Note: The facilities listed can only lower the outside humidity level. The facilities do not have the capability to raise outside humidity levels

Table 4.1.1 Environmental control specifications for the Eastern Range (ER)

4.1.3 Radiation and Electromagnetic Environments

The Delta II transmits on several frequencies to provide launch vehicle T/M and beacon signals to the appropriate range tracking stations. The Delta II also has uplink capability for command destruct receivers (CDR). The Delta II has three S-band T/M systems (one on each stage); three CDR systems, one on the first stage and two on the second stage; and a C-band transponder (beacon) on the second stage. Delta II RF systems are switched to internal power a few minutes before launch and remain on until battery depletion, except for the CDR system, which is commanded off in-flight.

On all three-stage missions, the upper stage incorporates a T/M package that may include an optional tracking beacon. In addition to the Delta II launch vehicle upper-stage instrumentation, the spacecraft usually incorporates special instrumentation to monitor spacecraft flight data on a spacecraft T/M channel, and downlinks this data to the appropriate range tracking station. The following paragraphs describe the T/M characteristics of each stage, the command destruct characteristics on the first and second stages, the second-stage C-band transponder characteristics, and upper-stage T/M instrumentation.

4.1.3.1 Launch Complex-17 Electromagnetic Environments

LC-17 electromagnetic environments are as follows:

10 kHz - 5.762 GHz	5 V/m
5.762 GHz - 5.768 GHz	40 V/m
5.768 GHz - 40 GHz	5 V/m

For radars under their control, the Western Range (WR) has techniques to control the SLC-2W pad environments to less than 5 V/m. As a minimum, the SP should run a radiated susceptibility test to insure survival at the above levels. Additionally, the SP must run a radiated emissions test per Figures 6-12 and 6-13 of MIL-STD-461C, RE-02.

4.1.3.2 Delta II First-Stage T/M Characteristics. Delta II first-stage T/M radiation characteristics are listed in Table 4.1.3. Radiation is continuous starting prior to launch and through first/second-stage separation.

4.1.3.3 Delta II Second-Stage T/M Characteristics. Delta II second-stage T/M radiation characteristics are listed in Table 4.1.3. Radiation is continuous starting prior to launch and through second/upper-stage separation and battery depletion.

4.1.3.4 Delta II Upper-Stage T/M Characteristics. Delta II upper-stage T/M radiation characteristics are listed in Table 4.1.3. Radiation is continuous from approximately 4 min prior to launch through loss of battery power. The RF-link utilizes a fairing-mounted antenna system prior to fairing jettison, and an upper-stage antenna system after fairing jettison.

4.1.3.5 Delta II First-Stage Command Destruct. Delta II first-stage CDR characteristics are listed in Table 4.1.3. For launch, the system is armed approximately 3 minutes prior to launch. Command destruct capability is maintained until 60 seconds after Delta II SECO 1.

4.1.3.6 Delta II Second-Stage Command Destruct. Delta II second-stage CDR characteristics (Table 4.1.3) are the same as those of the first stage.

4.1.3.7 Delta II Second-Stage C-Band Transponder. Delta II second-stage C-band beacon radiation characteristics are listed in Table 4.1.3. The tracking transponder is activated in terminal countdown and remains on throughout the flight.

4.1.3.8 Upper-Stage T/M Instrumentation. On all Delta II missions, a T/M system is installed on the upper stage to obtain performance data such as motor chamber pressure, acceleration, and battery voltage. Other spacecraft launch environment data such as low - and high-frequency vibration, acceleration transients, shock, velocity increments, indications of spacecraft separation, and temperature may also be obtained. The T/M transmitter operates at 2252.5 MHz with an output power ranging from 5 to 8 W. Modulation is FM/FM with standard IRIG subcarriers.

First-Stage T/M Radiation Characteristics

■ Transmitter	
● Nominal frequency	2244.5 MHz
● Power output	2.0 watts min 4.0 watts max
● Modulation Bandwidth	+ 550 kHz at 3 Db 1080 KHz at 60 Db
● Stability	+ 67 kHz max
■ Antenna	
● Type	Cavity-backed slot
● Polarization	Essentially linear parallel to booster roll axis
● Location	354 deg (looking aft) - Sta 1082 174 deg (looking aft) - Sta 1082
● Pattern	Nearly omnidirectional
● Gain	-2 dB min

Second-Stage T/M Radiation Characteristics

■ Transmitter	
● Nominal frequency	2241.5 MHz
● Power output	2.0 watts min 4.0 watts max
● Modulation Bandwidth	+ 550 kHz at 3 Db 1080 KHz at 60 Db
● Stability	+ 67 kHz max
■ Antenna	
● Type	Cavity-backed slot
● Polarization	Essentially linear parallel to booster roll axis
● Location	338 deg (looking aft) - Sta 559 151 deg (looking aft) - Sta 559
● Pattern	Nearly omnidirectional
● Gain	-2 dB min

Upper-Stage T/M Radiation Characteristics

■ Transmitter	
● Nominal frequency	2252.5 MHz
● Power output	5.0 watts min 8.0 watts max
● Modulation Bandwidth	+ 140 kHz at 3 Db 250 KHz at 60 Db
● Stability	+ 67 kHz max
■ Antenna	
● Type	Circumferential belt
● Polarization	Essentially linear parallel to booster roll axis
● Location	Belt at Sta 438
● Pattern	Nearly omnidirectional
● Gain	-2 dB min

Second-Stage C-Band Beacon Radiation Characteristics

■ Transmitter	
● Nominal frequency	5765 MHz (transmit) 5690 MHz (receive)
● Power output	440 watts min 1000 watts max
● Modulation Bandwidth	6 MHz at 6 Db
● Stability	3 MHz max
■ Antenna	
● Type	Transverse slot. Dipole loaded
● Polarization	Left hand circular
● Location	153 deg (looking aft) - Sta 559
● Pattern	Nearly omnidirectional
● Gain	+6 dB max

Command Destruct Receiver Radiation Characteristics

■ Receiver	
● Nominal frequency	416.5 MHz
■ Antenna	
● Type	Cavity-backed slot
● Polarization	Essentially linear parallel to booster roll axis
● Location	167 deg (looking aft) - Sta 559 347 deg (looking aft) - Sta 559 45 deg (looking aft) - Sta 695 225 deg (looking aft) - Sta 695
● Pattern	Nearly omnidirectional
● Gain	4.6 dB min

Table 4.1.3 Radiation Characteristics

4.1.4 Electrostatic Potential

The spacecraft must be equipped with an accessible ground attachment point to which a conventional alligator-clip ground strap can be attached. The vehicle/spacecraft interface provides the conductive path for grounding the spacecraft to the launch vehicle. Therefore, no dielectric coating is applied to the spacecraft interface. The electrical resistance of the spacecraft to vehicle interface surfaces must be 0.0025 ohms or less and is verified during spacecraft mate. (Reference MIL-B-5087B, Class R.)

4.1.5 Contamination and Cleanliness

The cleanliness conditions provided for the Delta II payloads represent minimum cleanliness conditions available. The following guidelines and practices from prelaunch through spacecraft separation, provide minimum Class 100,000 cleanliness (per Federal Standard 209B).

A. Precautions are taken during manufacture, assembly, test, and shipment to prevent contaminate accumulations in the Delta II upper-stage area, fairing, and PAF.

B. The encapsulation process is performed in a facility that is environmentally controlled to Class 100,000 conditions. All handling equipment is cleanroom-compatible and is cleaned and inspected before entering the facility. These environmentally controlled conditions are available for all remote encapsulation facilities and include Launch Complex (LC-17).

C. The nose fairing is cleaned using alcohol and then inspected for cleanliness prior to spacecraft encapsulation.

D. Personnel and operational controls are employed during spacecraft encapsulation to maintain spacecraft cleanliness.

For three-stage missions, contamination from spin rocket and Star 48 motor exhaust may be a concern. An optional plume shield, mounted on the second stage between the SP and the rockets, is available.

4.1.7 Winds Aloft Constraints

Measurements of winds aloft are taken at the space launch complex. The Delta II controls and loads constraints for winds aloft are evaluated on launch day by conducting a trajectory analysis using the measured wind. A curvefit to the wind data provides load relief in the trajectory analyses. The curvefit and other load relief parameters are used to reset the mission constants just prior to launch.

4.1.8 Lightning Activity

The following are procedures for test status during lightning activity:

A. Evacuation of the MST and fixed umbilical tower (FUT) is accomplished at the direction of the Test Conductor (TC) (Reference: Delta Launch Complex Safety Plan).
B. First - and second-stage instrumentation may be operated during an electrical storm.
C. If other vehicle electrical systems are powered when an electrical storm electro-explosive device (EED) circuits are electrically connected in the launch configurations, the GC must be turned off.
D. If an electrical storm passes through after a simulated flight test, all electrical systems are turned on in a quiescent state, and all data sources are evaluated for evidence of damage. This turn-on is done remotely (pad clear) if any category "A" ordnance circuits are connected for flight. Circuits are disconnected and safed prior to turn-on with personnel exposed to the vehicle.
E. If data from the quiescent turn-on reveals equipment discrepancies, which can be attributed to the electrical storm, a flight program requalification test must be run subsequent to the storm and prior to a launch attempt.
F. During terminal countdown, the launch director is responsible for initiating and ending an alert. Upon initiation of the alert, the GC is turned off. When the alert is lifted, the GC is turned on and memory verified.

4.2 LAUNCH AND FLIGHT ENVIRONMENTS

4.2.1 Fairing Internal Pressure Environment

As the Delta vehicle ascends through the atmosphere, venting of the fairing is provided by a 10 in^2 (64.5 cm^2) vent opening in the interstage and by other leak paths in the vehicle. The resulting internal pressure limits are presented in Figure 4.2.1 for the Delta 9.5-ft (2.9m) fairing. SPs are required to perform a venting analysis to demonstrate capability of surviving vehicle ascent.

4.2.2 Thermal Environment

During launch, the fairing and third stage contribute to the overall radiant heat input to the SP.

4.2.2.1 9.5ft Fairing. The thermal environment of the 9.5 foot fairing internal surfaces facing the spacecraft, unless modified by mission-specific requirements, is shown in Figure 4.2.2.1. These temperatures are based on typical depressed versions of the trajectory.

Fairing jettison will normally occur sometime after the theoretical free molecular heating for a flat plate normal to the free stream drops below 0.1 Bty.ft^2-sec (1135 W/m^2) based on the 1962 US Standard Atmosphere.

4.2.2.2 Upper-Stage Motor and Spin Rockets. For a three-stage mission the SP is exposed to plume heating from the spin rocket and third stage motor exhaust. This heat flux is a short duration pulse. An optional plume shield, using one of two existing designs, is available if required.

4.2.3 Flight Dynamic Environment

4.2.3.1 Maximum SP Steady-State Acceleration. Acceleration is a function of second stage payload (SP, primary payload, and upper stage) mass. For the Delta 7920 vehicle, the maximum axial acceleration typically occurs at the end of the first-stage burn.

4.2.3.2 Vibration Environment. The three vibration environments of concern to the user are sinusoidal, random, and acoustic. Each environment is discussed below.

Sinusoidal Vibration. The maximum expected sinusoidal vibration flight levels for the 7920/7925 vehicles are shown in Table 4.2.3.2. The flight level inputs account for vehicle mode shapes and changes expected from varying spacecraft dynamics. These inputs are defined at the interface between the longeron mounting brackets and the guidance section skin. The sinusoidal vibration levels cover transient and sinusoidal conditions that occur during flight.

Oscillations and Random Vibration. The SP random vibration environment is created by the acoustic noise generated at liftoff and during transonic flight. The random vibration interface levels of Figure 4.2.3.2a and 4.2.3.2b are representative of random vibration levels for separating and non-separating SPs, respectively.

4.2.3.3 Acoustic Environment. The highest acoustic environment occurs at liftoff and during transonic flight.

The SP acoustic environment for the 9.5-foot fairing is shown in Figure 4.2.3.3. The environment for the 10-foot fairing is slightly lower.

4.2.3.4 Shock Environment. Maximum shock response for separating SPs occurs during spacecraft/Delta separation. Delta vehicle-induced high-frequency transients during fairing separation and second-stage separation are the dominant shock environments for non-separating SPs.

For separating SPs the PAF uses a v-block clamp separation system. Separation is achieved by cutting the two bolts that hold the v-block clamp assembly together. The maximum anticipated spacecraft separation shock spectrum is presented in Figure 4.2.3.4a. This criterion represents the shock level at the spacecraft/PAF separation plane. Typical of this type of shock, the shock level dissipates rapidly with distance and the number of joints between the explosive cutter and the point of measurement. These levels are dependent on interface geometry and preloads. Specific interface shock levels are derived for each spacecraft when the interface configuration is defined.

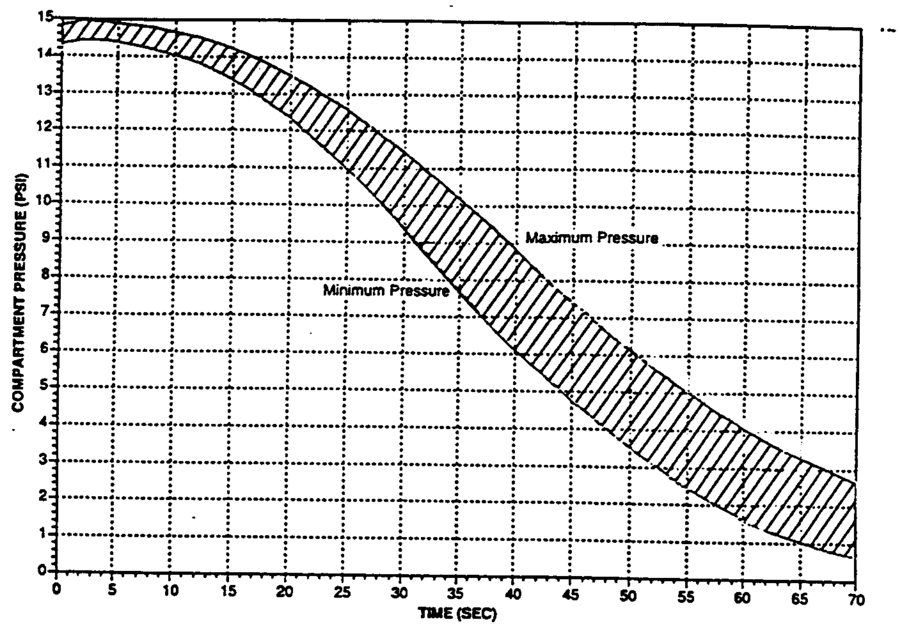


Figure 4.2.1 9.5-Foot (2.9 m) Fairing Internal Pressure Limits

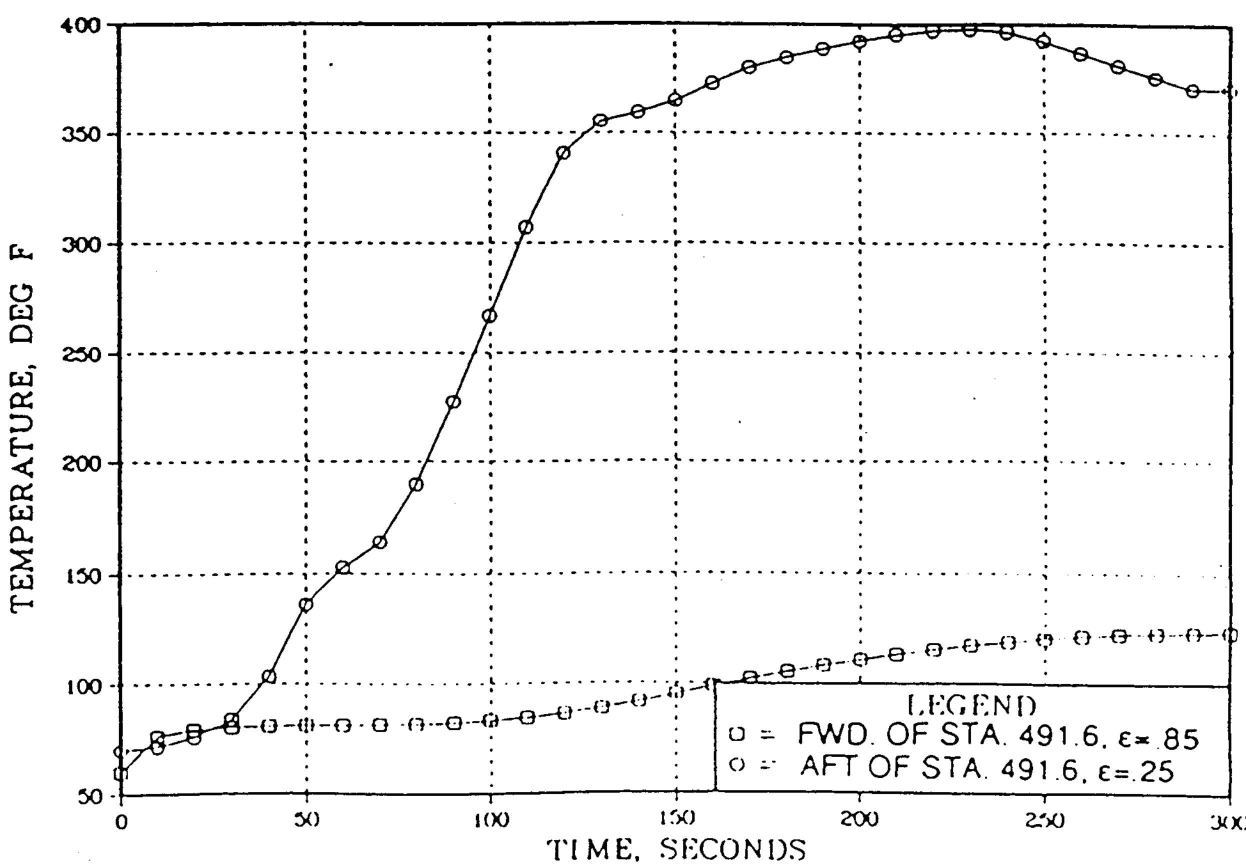


Figure 4.2.2.1 9.5-Foot Fairing Aft Cylinder Internal Wall Temperature and Emittance

MAXIMUM FLIGHT LEVEL		PROTOFLIGHT TEST LEVEL	
Frequency (Hz)	Level (Go-p)	Frequency (Hz)	Level G(o-p) (Max Flight + 3 dB)
Thrust Axis (Delta Vehicle)		Thrust Axis (Delta Vehicle)	
5 - 6.2	0.5 inch D. A.	5 - 7.4	0.5 inch D. A.
6.2 - 100	1.0	7.4 - 100	1.4
Radial and Tangential Axes (Delta Vehicle)		Radial and Tangential Axes (Delta Vehicle)	
5 - 100	0.7	5 - 6.2	0.5 inch D. A.
		6.2 - 100	1.0
Sweep Rate = 4 Octaves/Minute		Sweep Rate = 4 Octaves/Minute	

D. A. = Double Amplitude

Table 4.2.3.2 Sinusoidal Vibration Levels

Frequency (Hz)	Maximum Flight Level		Protoflight Test Level (Max Flight + 3dB	
10 - 20	0.001	G^2/Hz	0.002	G^2/Hz
20 - 60	+5.7	dB/Octave	+5.7	dB/Octave
60 - 220	0.008	G^2/Hz	0.016	G^2/Hz
220 - 400	+15.7	dB/Octave	+15.7	dB/Octave
400 - 700	0.18	G^2/Hz	0.36	G^2/Hz
700 - 900	-15.4	dB/Octave	-15.4	dB/Octave
900 - 1300	0.05	G^2/Hz	0.10	G^2/Hz
1300 - 1500	+12.4	dB/Octave	+12.4	dB/Octave
1500 - 2000	0.09	G^2/Hz	0.18	G^2/Hz
Overall Grms =	12.9		18.2	
Duration =	30 Seconds/Axis 3 Axes		60 Seconds/Axis 3 Axes	

MAXIMUM FLIGHT LEVEL
RANDOM VIBRATION

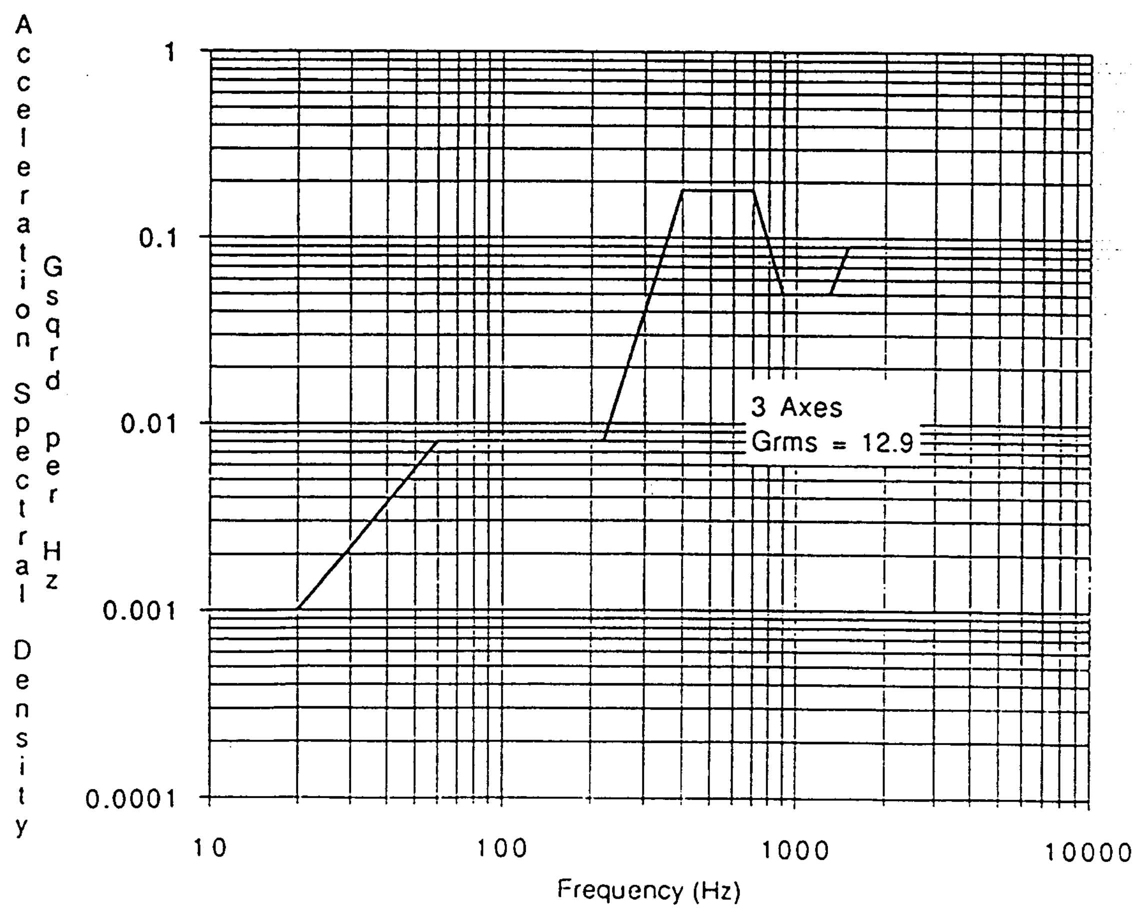
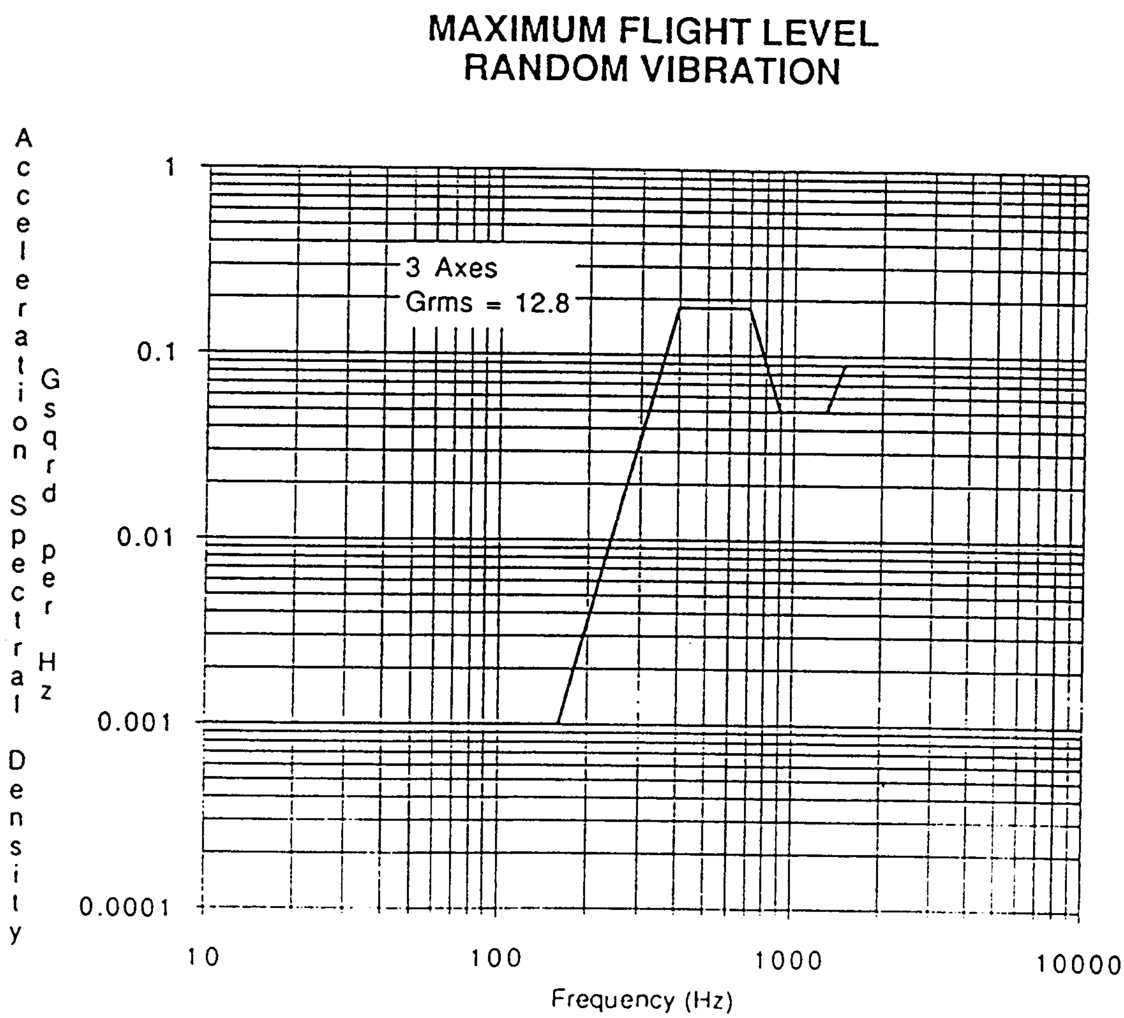


Figure 4.2.3.2a Maximum Flight Level Random Vibration for Separating SPs

Note: Level is defined at Delta to SP separation plane

Frequency (Hz)	Maximum Flight Level	Protoflight Test Level (Max Flight +3 dB)
10 - 160	0.001 G^2/Hz	0.002 G^2/Hz
160 - 400	+17.1 dB/Octave	+17.1 dB/Octave
400 - 700	0.18 G^2/Hz	0.36 G^2/Hz
700 - 900	-15.4 dB/Octave	-15.4 dB/Octaye
900 - 1300	0.05 G^2/Hz	0.10 G^2/Hz
1300 - 1500	+12.4 dB/Octave	+12.4 dB/Octave
1500 - 2000	0.09 G^2/Hz	0.18 G^2/Hz
Overall Grms =	12.8	18.1
Duration =	30 Seconds/Axis 3 Axes	60 Seconds/Axis 3 Axes



Note: Level is defined at guidance section skin/longeron interface.

Figure 4.2.3.2b Maximum Flight Level Random Vibration for Non-Separating SPs

Frequency (Hz)	Maximum Flight Level (dB)
31.5	116.0
40	120.0
50	123.0
63	127.0
80	128.5
100	128.5
125	128.5
160	128.5
200	129.5
250	132.0
315	134.5
400	139.0
500	141.0
630	139.0
800	131.0
1000	129.0
1250	128.5
1600	127.5
2000	127.5
2500	124.5
3150	121.5
4000	118.0
5000	117.0
6300	114.0
8000	113.5
10000	113.5

OASPL = 146.2 dB
Duration = 30 Seconds

MAXIMUM FLIGHT LEVEL

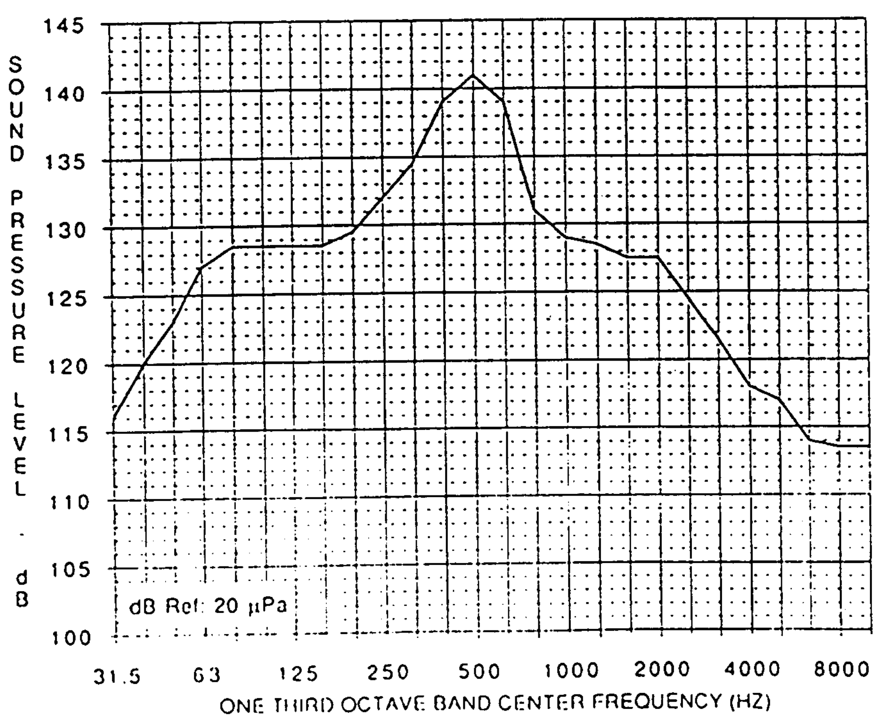


Figure 4.2.3.3 9.5-Foot (2.9 m) Fairing Internal Acoustic Environment for the 7925 Vehicle

For non-separating SPs, the spacecraft interface shock environment is provided in Figure 4.2.3.4b. The environment is defined at the spacecraft-to-longeron interface plane.

4.2.3.5 Combined Loads. Delta flight dynamic excitations are superimposed on steady-state accelerations to produce combined accelerations that must be used in the spacecraft structural design. The combined spacecraft accelerations are a function of launch vehicle characteristics, the SP mass and dynamic characteristics, and the primary spacecraft mass and dynamic characteristics. The SP project provides a preliminary math model and then a model correlated to modal test results. Boeing determines coupled modes of the SP and the second stage. **In general, if coupled modes are greater than 50 Hz, a coupled loads analysis is not required.** Final determination is made on a case by case basis. Load factors are determined using the coupled mode frequencies. For preliminary structural design of SPs with coupled modal frequencies greater than 35 Hz, a value of 10 g's limit should be used acting at the SP center of gravity in three axes simultaneously. For SPs, or separate components of the SP, that are less than approximately 11 kg (25 lbs), preliminary design should use 20 g's limit at the center of gravity in three axes simultaneously. These values are confirmed by Boeing review and coordination of the SP dynamic characteristics.

4.2.4 Flight Acceptance and Qualification Test Levels

This paragraph outlines a series of flight acceptance and qualification environmental test requirements for spacecraft launched on Delta II vehicles. Protoflight level testing may be the most cost effective method for one-of-a-kind payloads. It is important to note that PS testing not only assures survival of the launch, but also demonstrates that there is no impact to the primary mission.

The flight acceptance test levels established subject a flight spacecraft to its maximum flight environments. The qualification test levels established are intended to assure through reasonable overtest that the spacecraft, even with monor weight and design variations, can withstand the most severe Delta dynamic environmental loads. Once a spacecraft test unit has been subjected to qualification levels, flight units are tested at flight acceptance levels. Protoflight testing is intended for a flight spacecraft which has not had qualification testing on a dedicated test unit. Protoflight tests should be performed in the final flight configuration with minimal disassembly permitted after the test. Test components that are flight configuration may be used in some cases, e.g. batteries. **A detailed list of all deviations from the flight configuration and all planned SP disassembly is required in the test plan.** These items will be reviewed and approved by NASA/KSC, Boeing and USAF.

4.2.4.1 Structural Load Tests/Analyses. Structural load testing is performed by the user to demonstrate the design integrity of the primary structural elements of the spacecraft. These loads are on the basis as defined in Section 4.2.3.5. Maximum flight loads will be increased by a factor of 1.25 to determine qualification test loads. In lieu of structural load testing, analysis is an acceptable means of providing flight verification by the use of a "no test" factor of 2.0 times the maximum flight levels. Untested structure shall use a 1.65 factor of safety for yield analysis.

4.2.4.2 Sinusoidal Vibration Tests. The flight levels in Section 4.2.3.2 are multiplied by a 1.4 factor for spacecraft protoflight testing. This factor is intended to ensure a reasonable degree of overtest and to provide contingencies for such factors as (1) limited flight data and (2) test tolerances (equipment and operation). The sinusoidal protoflight levels provided are applied at the interface between the longeron mounting bracket and the guidance section skin.

The protoflight tests include one sweep through the given frequency range at the rate of four octaves per minute. Flight acceptance test levels, also provided in Table 4.2.3.2, include one sweep through the given frequency range at the rate of four octaves per minute. The tolerance on amplitude input shall not exceed $\pm$ 10% for all tests. Sinusoidal vibration accelerometer data shall be presented as peak acceleration (Gpk) versus frequency (Hz) from 5 to 100 Hz.

Frequency (Hz)	Maximum Flight Shock Response Spectrum Level (Q=10)
100 - 140	50 G
140 - 300	+12.7 dB/Octave
300 - 500	250 G
500 - 6000	+6.7 dB/Octave
6000 - 7000	4000 G
7000 - 10000	-1.3 dB/Octave
10000	3700 G

1 Shock/Axis
3 Mutually Perpendicular Axes

MAXIMUM FLIGHT LEVEL SHOCK RESPONSE SPECTRUM

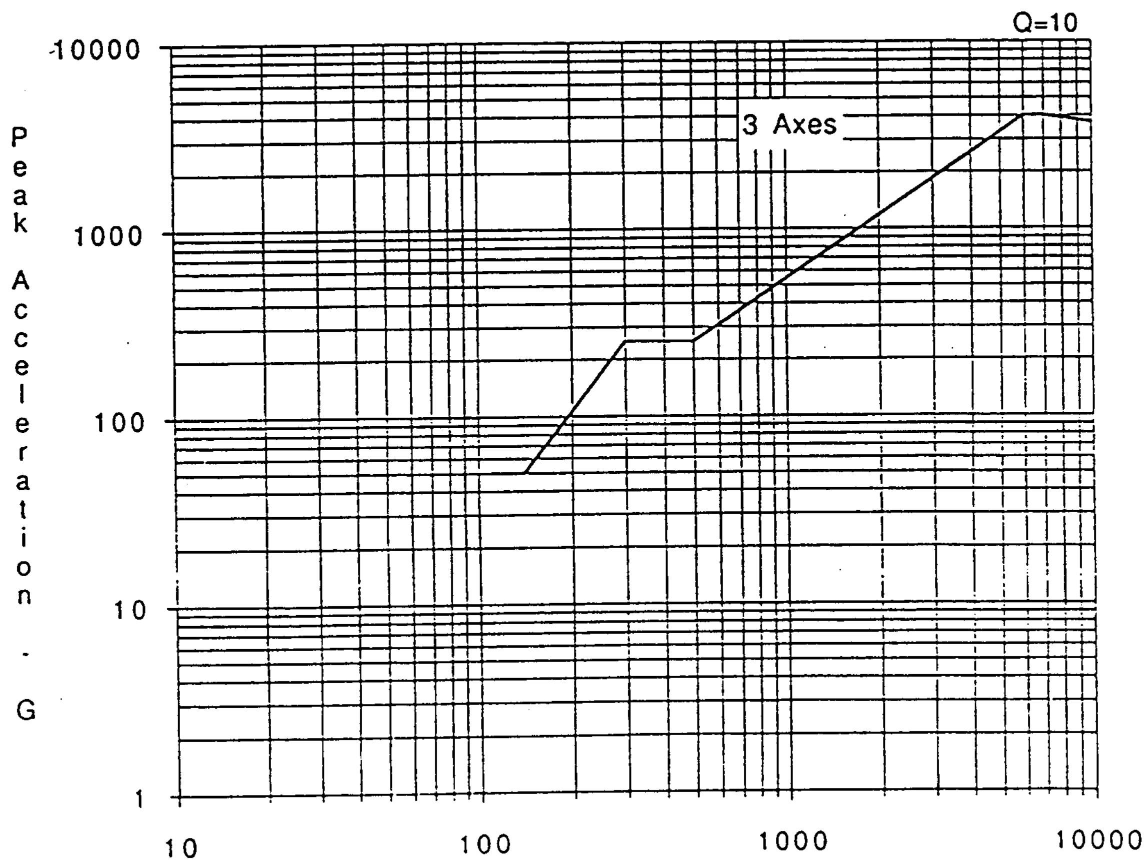


Figure 4.2.3.4a Maximum Flight Level Shock Response Spectrum for Separating SPs

Note: Level is defined at Delta to SP separation plane.

Frequency (Hz)	Maximum Flight Shock Response Spectrum Level (Q=10)	Protoflight Test Level (Max Flight +3 dB)
100	9 G	13 G
100 - 1000	+12.0 dB/Octave	+12.0 dB/Octave
1000 - 1200	900 G	1260 G
1200 - 1500	-7.6 dB/Octave	-7.6 dB/Octave
1500 - 10000	680 G	960 G
	2 Shocks/Axis 3 Axes	2 Shocks/Axis 3 Axes

MAXIMUM FLIGHT LEVEL
SHOCK RESPONSE SPECTRUM

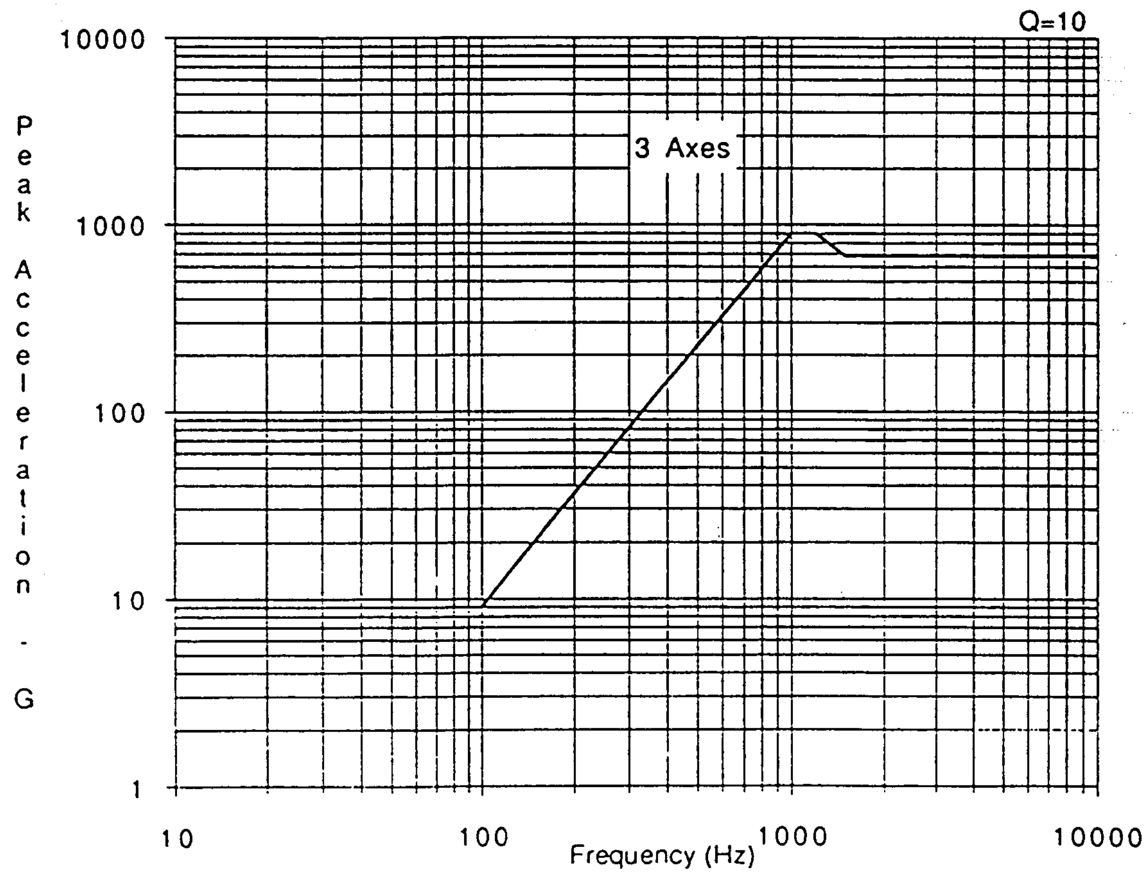


Figure 4.2.3.4b Maximum Flight Level Shock Response Spectrum for Non-Separating SPs

Note: Level is defined on SP support longerons.

4.2.4.3 Acoustic Noise Tests. The flight levels defined in Section 4.2.3.3 are increased by 3 dB for protoflight tests. The tolerances are +4 dB and -2 dB from 50 Hz to 2000 Hz. Above and below those frequencies the levels are maintained as close to the nominal values as possible within the limitations of the test facility. The overall sound pressure level (OASPL) should be maintained within +3 dB and -1 dB of the requirement.

4.2.4.4 Random Vibration Tests. For payloads that do not have large flexible surface areas (e.g. solar arrays), random vibration tests may be used in lieu of acoustic tests. The flight and protoflight levels are given in Figures 4.2.3.2a and 4.2.3.2b.

The RMS test level tolerance shall be ± 10 percent. The acceleration spectral density tolerance shall be ± 1.5 dB when measured with a 25 Hz or narrower bandpass filter from 10 to 500 Hz, and ± 2.0 dB when measured with a 50 Hz or narrower filter from 500 to 2000 Hz, with the mean value of the variations remaining approximately equal to the nominal value. Plus/minus 4 dB shall be permitted for any narrow band spectral excess (spike) whose bandwidth at the appropriate tolerance level is less than 10 Hz or less than 10 percent of the geometric mean frequency where the spike occurs, whichever is greater.

4.2.4.5 Shock Tests. Pyrotechnic shock levels for separating SPs are difficult to simulate mechanically on a complete spacecraft without severe overtest in the low-frequency region. The more direct method is to use actual Delta II ordnance devices in a test on the flight-type attach fitting. Performing such a test twice qualifies a spacecraft to withstand the flight shock environment. Protoflight testing also requires two test. For flight acceptance, the test shall be performed once. Test hardware is made available through the NASA/KSC/ELV Project.

For non-separating payloads, the shock requirements are shown in Figure 4.2.3.4b.

Shock test accelerometer data shall be reduced as shock response spectra showing peak acceleration response (G's) versus frequency (Hz). The data shall be analyzed with a damping ratio of 0.05 (Q = 10) using 1/6 or narrower octave band filters. The accelerometer data shall also be presented as acceleration time histories showing acceleration (G's) versus time (milliseconds).

The positive and negative shock response spectrum (SRS) of the transient excitation shall be within +4 dB and -2 dB of the required SRS from 100 to 3000 Hz. The SRS shall be calculated or analyzed with a Q of 10 (damping ratio of 0.05) using 1/6 or narrower octave band filters.

4.2.4.6 Modal Testing. As noted in Section 4.2.3.5, modal testing is required to verify the dynamic model used to determine SP and vehicle loads. For SP's with sufficient stiffness or of low enough mass, model testing may be minimized or eliminated. Modal test requirements are addressed early in the program. The dynamic modeling, modal testing, and coupled loads analysis process is typically the critical path for SP analysis. SP programs should have a preliminary model ready for delivery as early as possible in the program. It is strongly suggested that the model be in MSC NASTRAN or the Craig Bampton format.

4.2.5 Dynamic analysis Criteria and Balance Requirements

Two-stage primary payloads and SP mission may utilize the capability of the second stage to provide the terminal velocity, roll, final spacecraft orientation, and separation.

For some missions, the primary spacecraft or SP may require a low roll rate during coast or at separation. The Delta second stage can provide roll-up by commanding a roll rate using control jets. Roll rates up to 4 rpm (0.42 rad/s) can be accomplished. Higher roll rates are also possible; however, the accuracy is degraded as the rate increases.

For separating SPs, there is no dynamic balance requirement, but the static imbalance directly influences the spacecraft angular rates at separation. When there is a separation tipoff requirement, the SP CG offset must be reduced as much as possible. Typical mean tipoff rates increase by about 2 deg/sec per 0.025 cm (0.01 in) CG offset; three sigma rates are higher.

Section 5
Interface Hardware and Secondary Payload Envelopes

Detailed descriptions and requirements of launch vehicle mechanical and electrical interfaces with the spacecraft are presented in this section. The user ,,- shall adhere to the interface requirements and restraints. **Waivers are granted when necessary or appropriate, but all deviations or exceptions must be coordinated with and approved by NASA/KSC/ELV Launch Services and Boeing before they are implemented.**

5.1 SP ACCOMMODATION

Because of the development time and cost associated with custom interface hardware, it is to the advantage of the SP agency to use the existing designs. As early as possible in the spacecraft design phase, selection of an appropriate interface should be coordinated with NASA/KSC/ELV Launch Services.

All SPs flown to date have been mounted in the available space between the fairing and the second stage (see Figures 3-la and 3.1b). The SP usable envelopes are described in Section 5.7.1.

5.2 INTERFACE CONSTRAINTS

Of the SP mission constraints noted in the Overview, the following relate to the hardware interface and envelope:

o The SP may not intrude on the primary payload clearance envelope. Existing mounting hardware designs accommodate a variety of sizes and shapes of SPs. However, in general, one dimension of the SP must be less than 31 cm (12 in) for separable SPs or 38 cm (15 in) for non-separating SPs. Envelopes are about 0.5 in larger for NASA missions.

o SPs shall incorporate dedicated power, sequencing, and wiring isolated from the primary payload. Support components may be provided by the vehicle contractor, The Boeing Company (Boeing).

o New/revised structure to accommodate SPs shall be designed to an ultimate factor of safety of 2.00 if not tested or 1.25 if tested.

0 The support structure, clamp bands, and separation systems for SPs will be provided by Boeing or be of proven design and approved by NASA/KSC/ELV Launch Services.

o SP telemetry can be routed through the Delta II antenna and shall have separate verification.

o Support from the second stage, such as power, pointing, or separation command, is generally limited to a mission duration of about one orbit revolution unless extra-cost modifications are provided to extend support for several revolutions. Details on limitations on mission duration are provided in Section 2.1.

5.4 THE PAF ASSEMBLY FOR SEPARATING PAYLOADS

Two PAFs have been developed and flown with SPs. Figure 5.5 shows the light-weight version. The Payload Adapter Assembly (PAA) is fastened to the PAF by a two-piece v-block type clamp assembly which is secured by two studs. These clamp bands are approximately 9 inches in diameter. The following figures and sections address the light-weight clampband, which is the most commonly used. The heavy-weight clampband will only be used on an exception basis.

Spacecraft separation is affected by actuation of ordnance cutters that sever the two studs. Clamp assembly design is such that cutting either stud will permit spacecraft separation. For the light weight clamp, springs assist in retracting the clamp assembly into retainers after release. A relative separation velocity of about 2 to 8 ft/sec (0.6 to 2.4 m/s) is imparted to the spacecraft by four separation springs. An example of a payload interface adapter is shown in Figure 5.5a.

Boeing supplies the PAF. The PAA is supplied by the SP project. Operations normally require that the PAF and the PAA be assembled with the clamp band prior to spacecraft installation. The entire assembly is then mated to the launch vehicle. Therefore, the PAA should be removable from the SP. Special provisions are required of the SP if the PAA is not removable. A previously flown PAA is shown in Figures 5.5a, b, and c. Deviations should be coordinated with NASA/KSC/ELV Launch Services.

Provisions on the PAA for separation switches are available. Two switch mounts can be installed into the PAA to accommodate SP-supplied switches. These switches can be long-lead items; therefore, proper advanced planning is recommended.

5.6 THE INTERFACE HARDWARE FOR NON-SEPARATING PAYLOADS

Four existing designs are available for non-separating SP interface hardware. These are shown in Figures 5.6.1 through 5.6.4 (tolerances per ANSI Y14.5M-1982).

5.7 ENVELOPES FOR SECONDARY PAYLOADS

Envelopes available for SPs depend on the fairing diameter and the primary spacecraft PAF for the assigned mission.

5.7.1 9.5-Foot Fairing Missions

Envelopes for two possible layouts utilizing the SP separation system are shown in Figures 5.7.1.la through 5.7.1-le and 5.7.1.2. For reference, these layouts are designated Separating Payload Envelope 1 (SPEL) and SPE2. Figure 5.7.1.1 shows available space for SPEI for launches on two- **or** three-stage missions with the various primary payload attach fittings. SPE2 is not available for two-stage missions because of PAF interference.

Envelopes for non-separating SPs can be visualized from those for separating SPs by noting that all space above the attach longerons, including the space taken by the PAF for separating SPs, is available. An example of possible utilization of space for a non-separating SP, in this case a tether deployer used for the Small Expendable Deployer System (SEDS) mission, is shown in Figures 5.7.1.3a. Figures 5.7.1.3b through 5.7.1.3d show the envelopes for non-separating SPs with interfaces shown in Figures 5.6.1 through 5.6.4. Figure 5.7.1.3e shows an end view of separating and non-separating SPs side by side in the space available between the Delta guidance compartment access doors, which must have clear access.

5.7.2 10-Foot Fairing Missions

The 10-foot fairing is used primarily for two-stage missions. The envelope forl, SPs utilizing the 10-foot fairing and the mounting of Figure **5.7.1.1** (adapter' centerline normal to vehicle centerline) is shown in Figure 5.7.2.

5.8 Structural Capability

Figure 5.8 provides an envelope of allowable center of gravity and weights for the lightweight and heavyweight versions of the clampband system. The envelope is intended for preliminary design only and is subject to verification by coupled loads analysis.

Non-separating SPs up to 154 lbs and a center of gravity of up to 7 inches from the interface have been flown on SPI3. Other interfaces for non-separating payloads have a lower capability. As with separating payloads, this information is provided for preliminary design, and is subject to verification by coupled loads analysis.

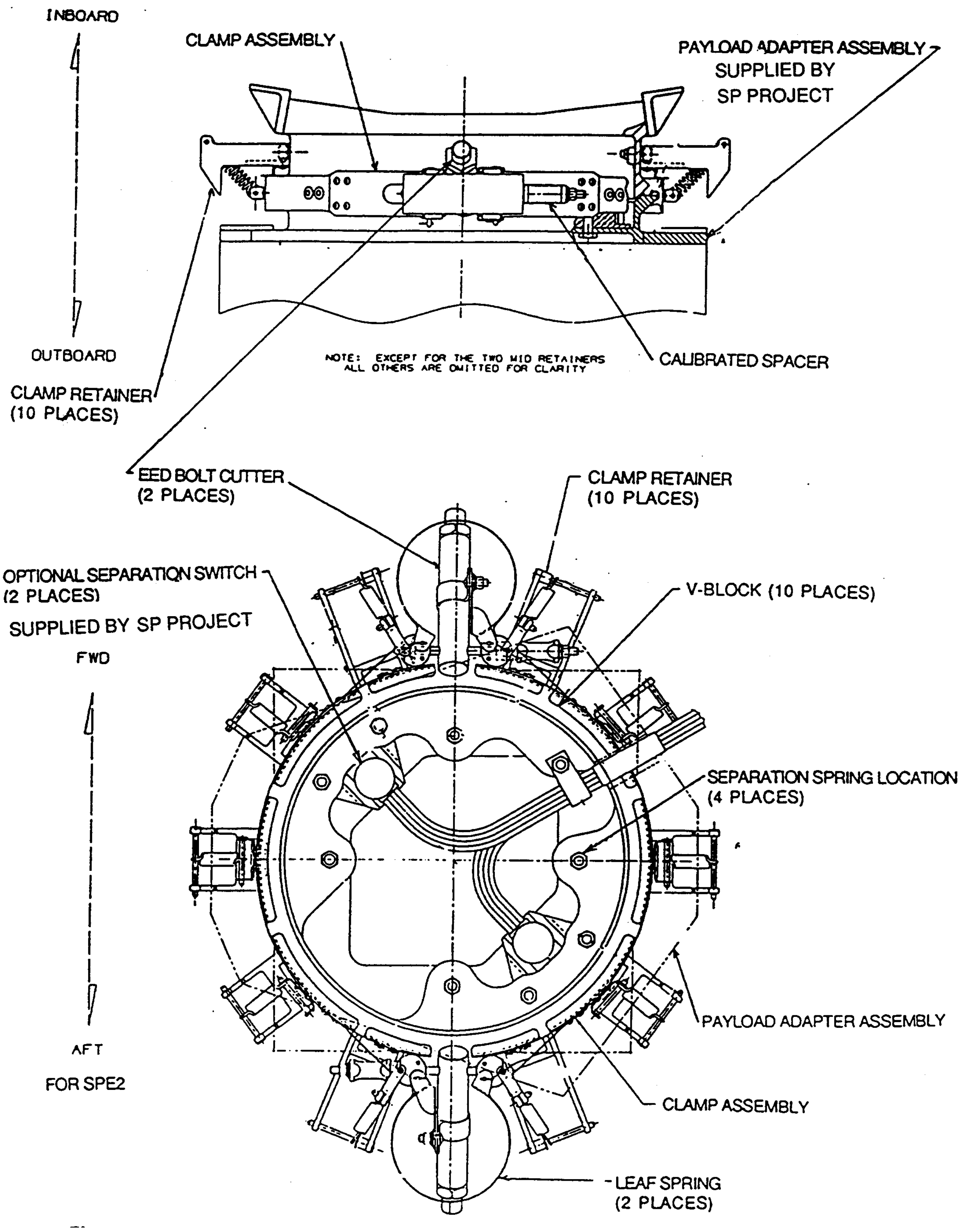


Figure 5.5 9 in. PAF for Separating SPs

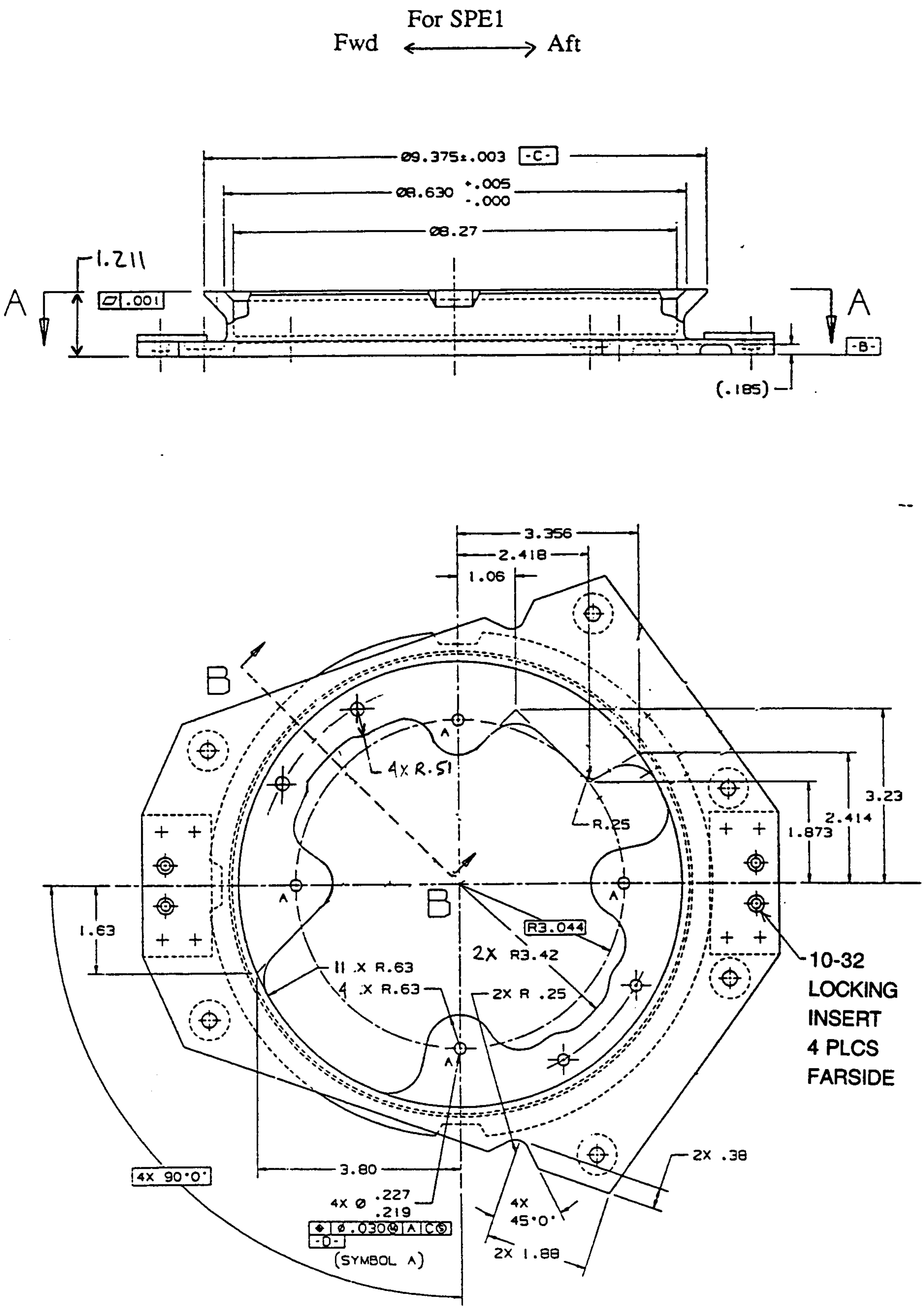


Figure 5.5a Example of Payload Adapter Assembly for Separating SPs

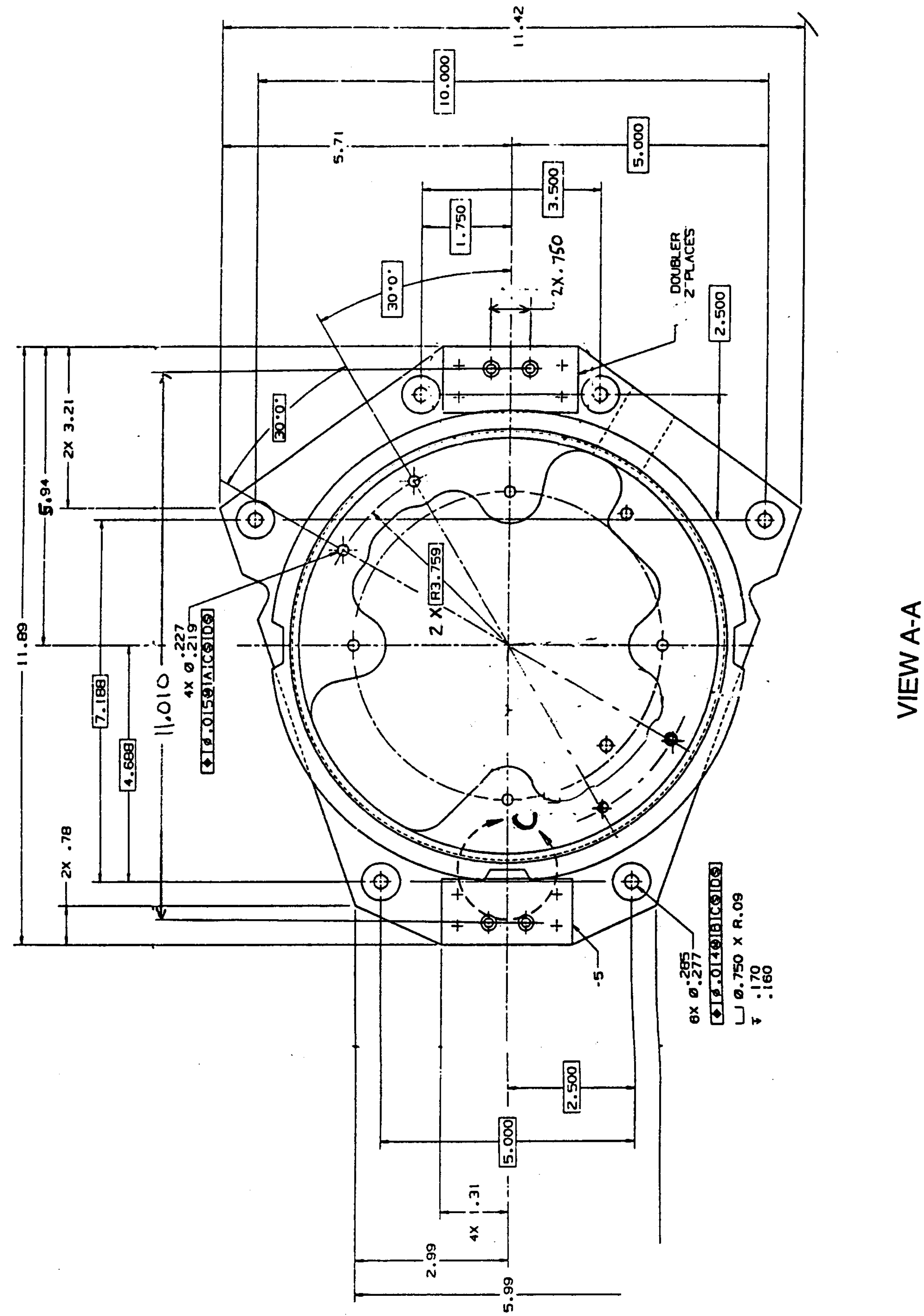
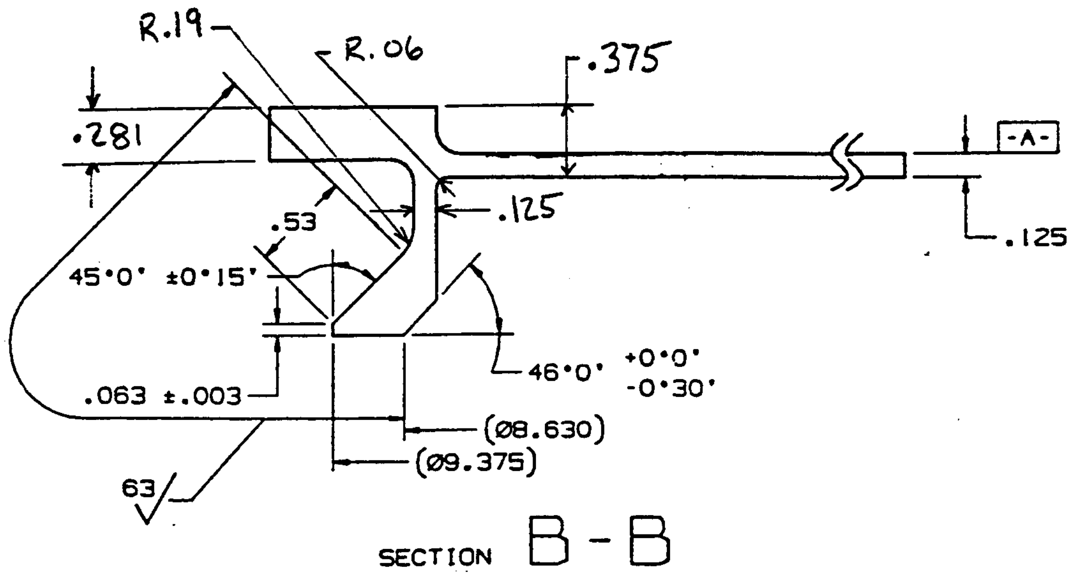


Figure 5.5b Example of Payload Adapter Assembly for Separating SPs



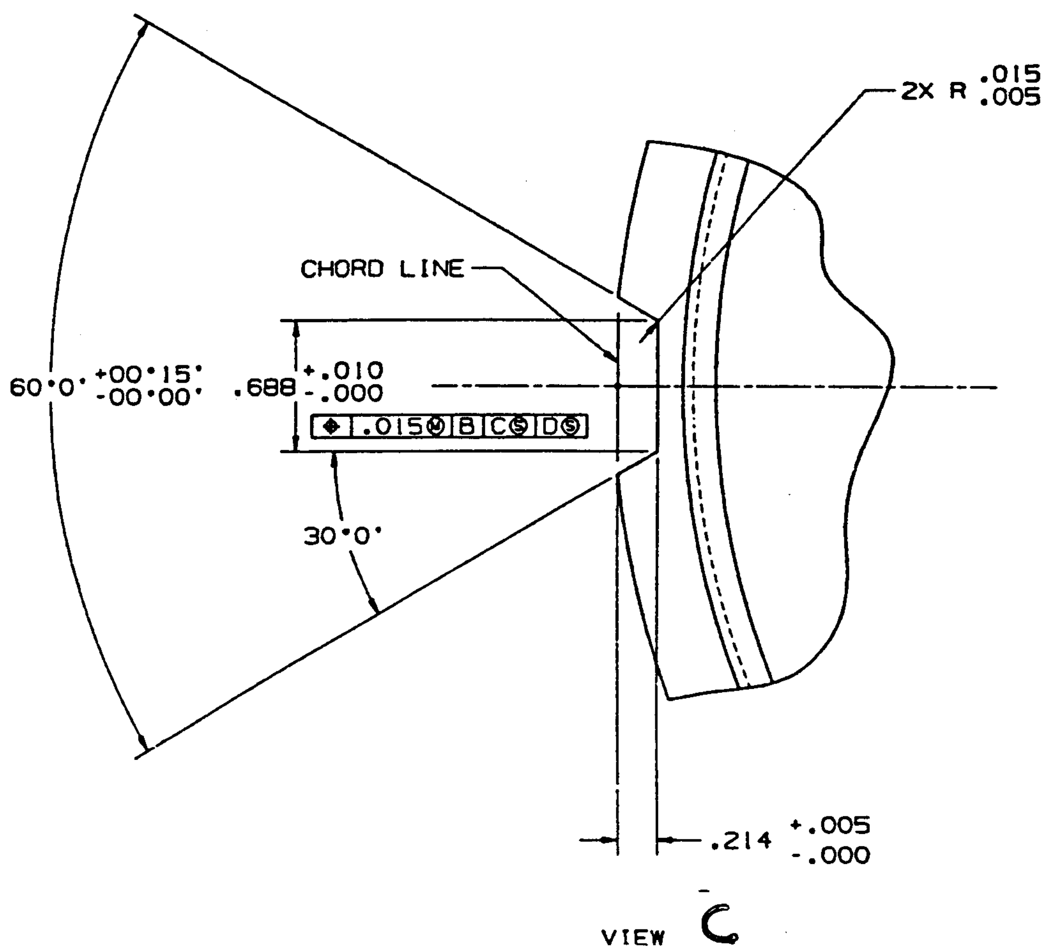


Figure 5.5c Example of Payload Adapter Assembly for Separating SPs

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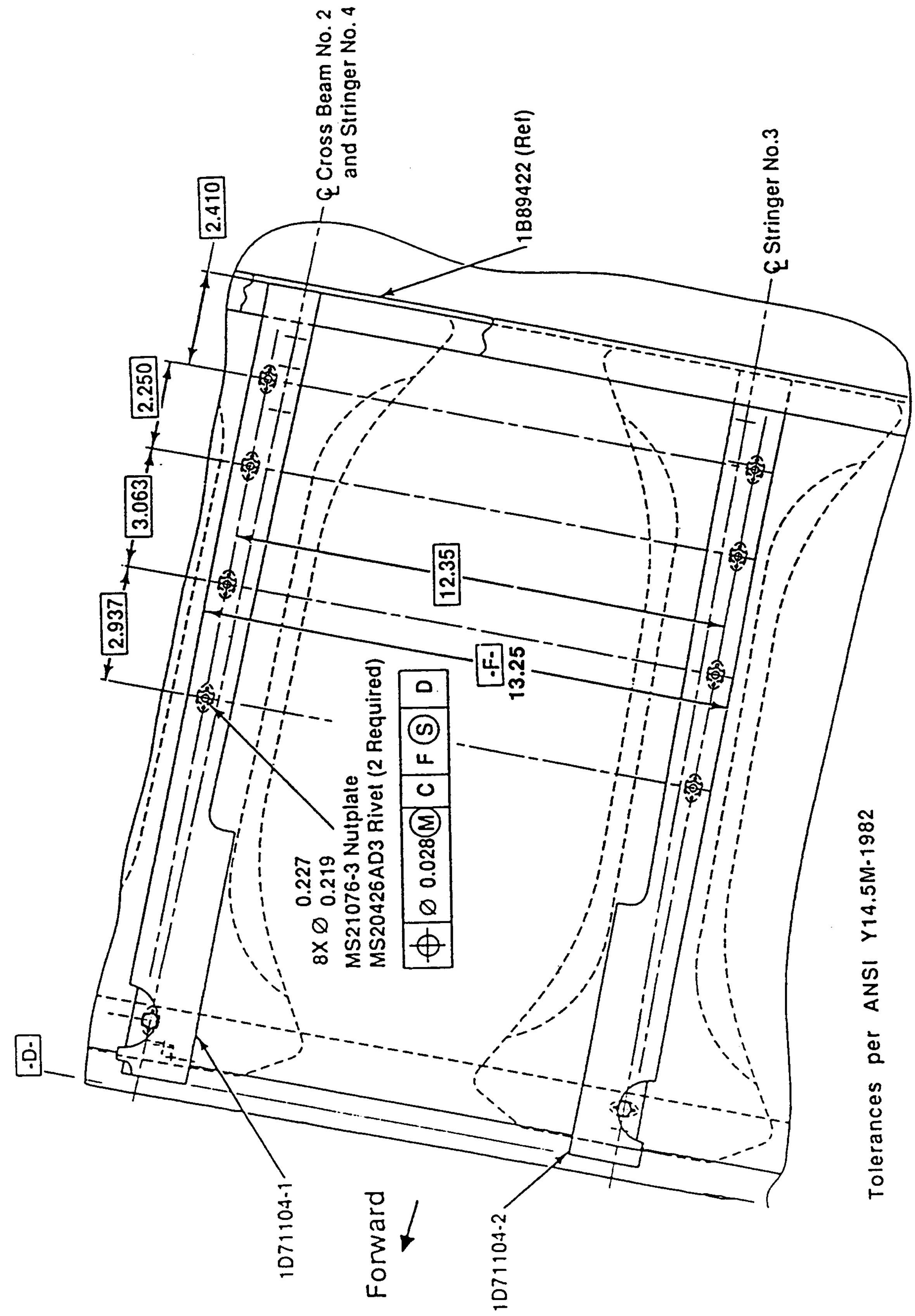


Figure 5.6.1 Non-Separating SP Interface SPI1

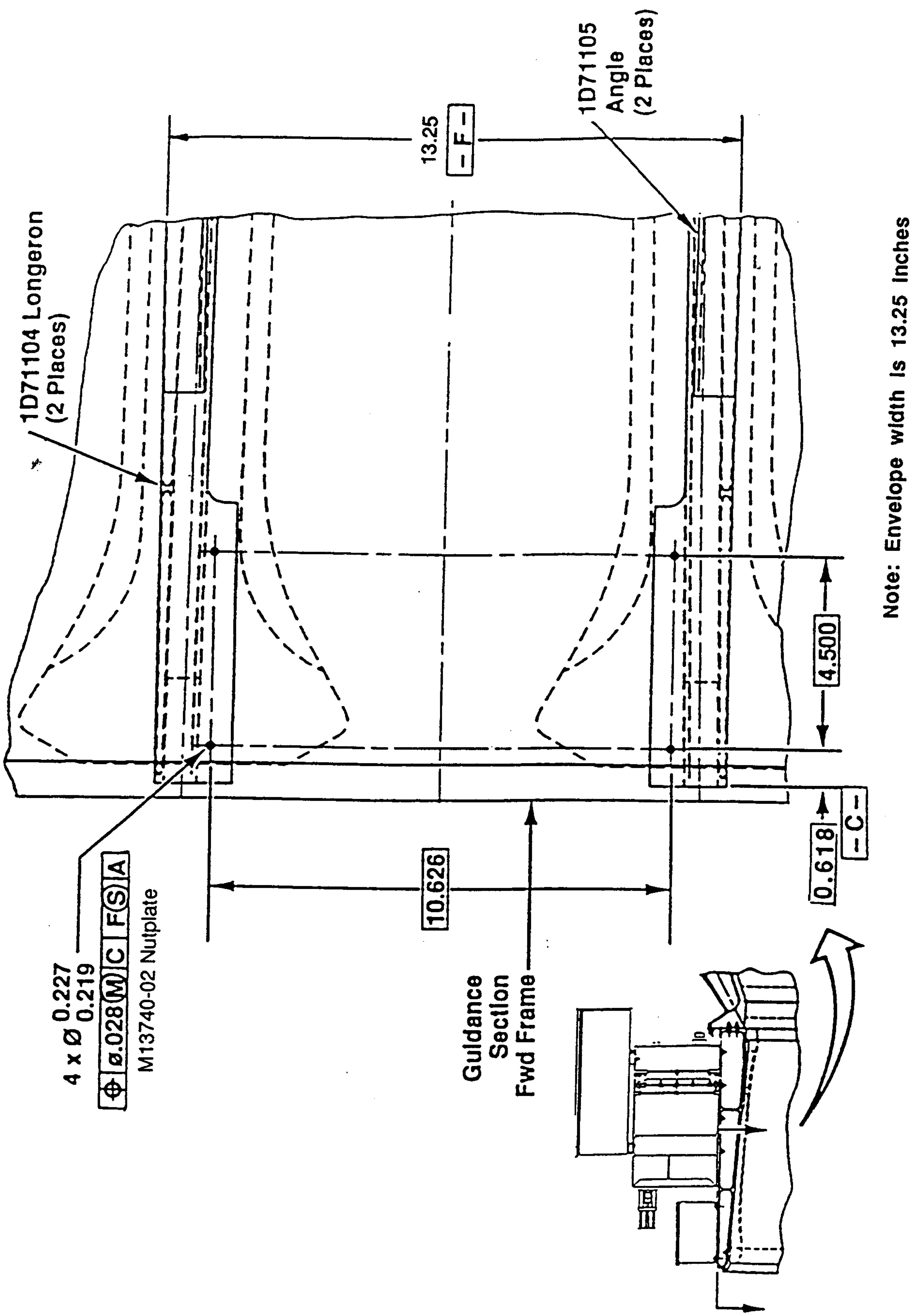


Figure 5.6.2 Non-Separating SP Interface SPI2

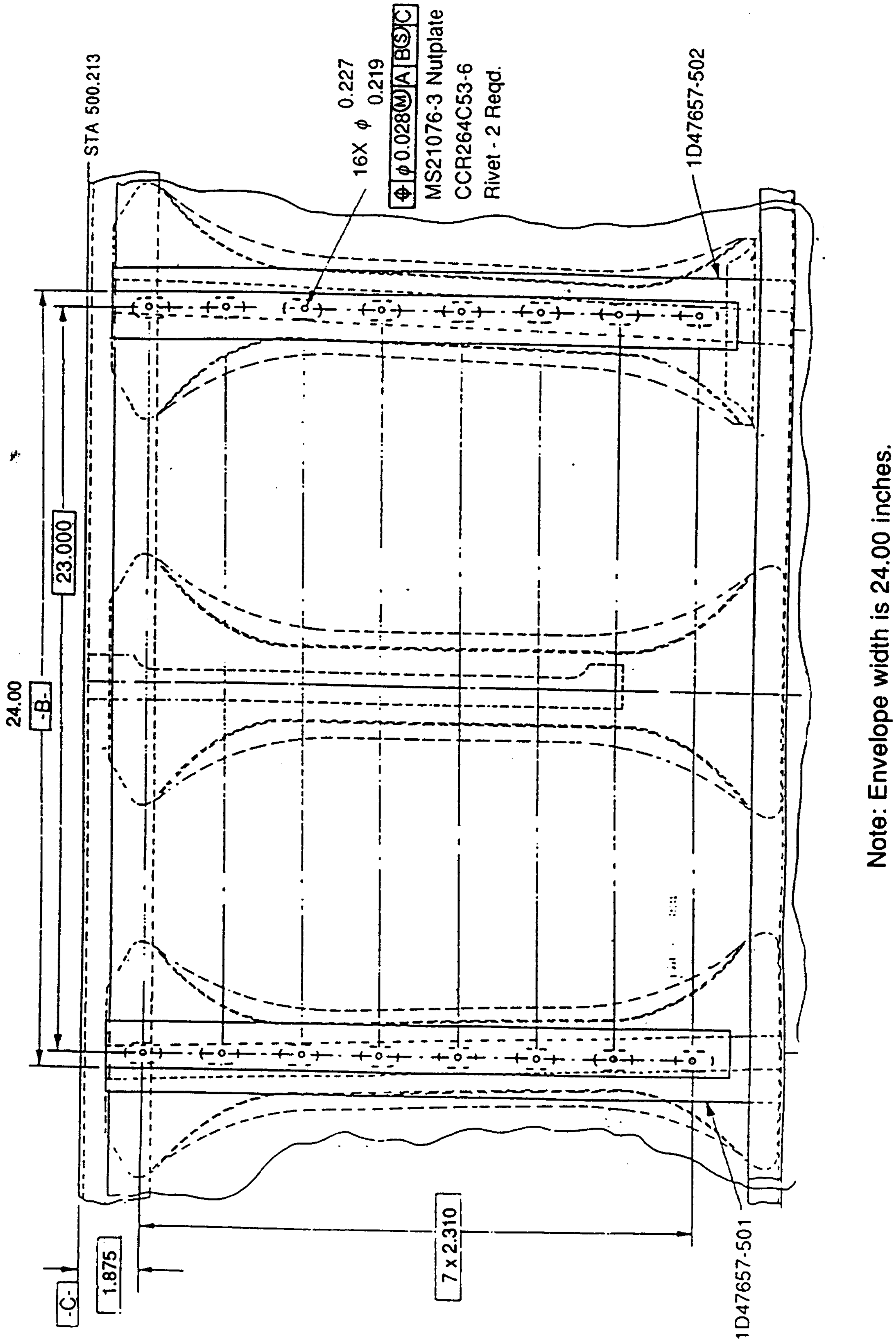


Figure 5.6.3 Non-Separating SP Interface SPI3

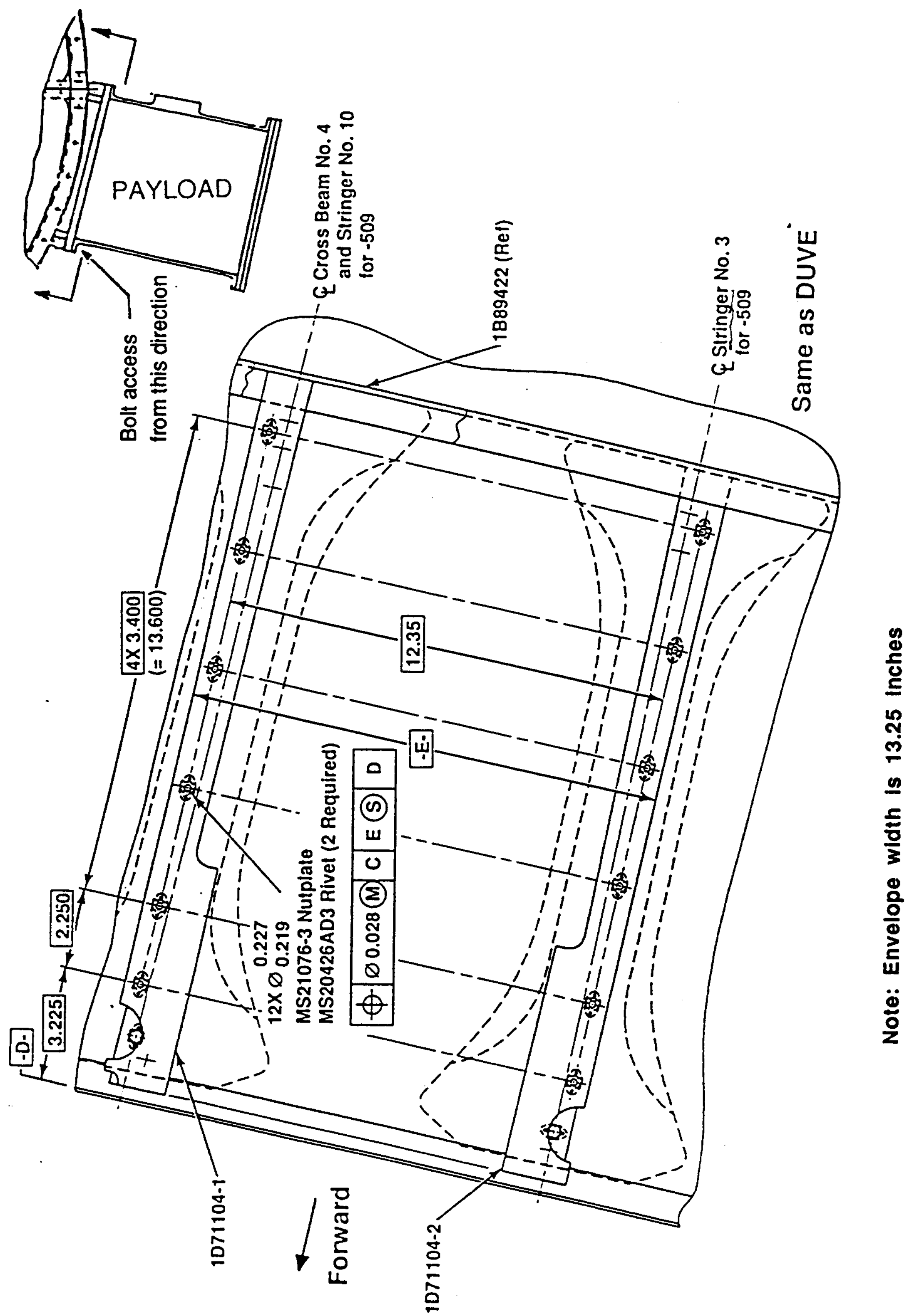


Figure 5.6.4 Non-Separating SP Interface SPI4

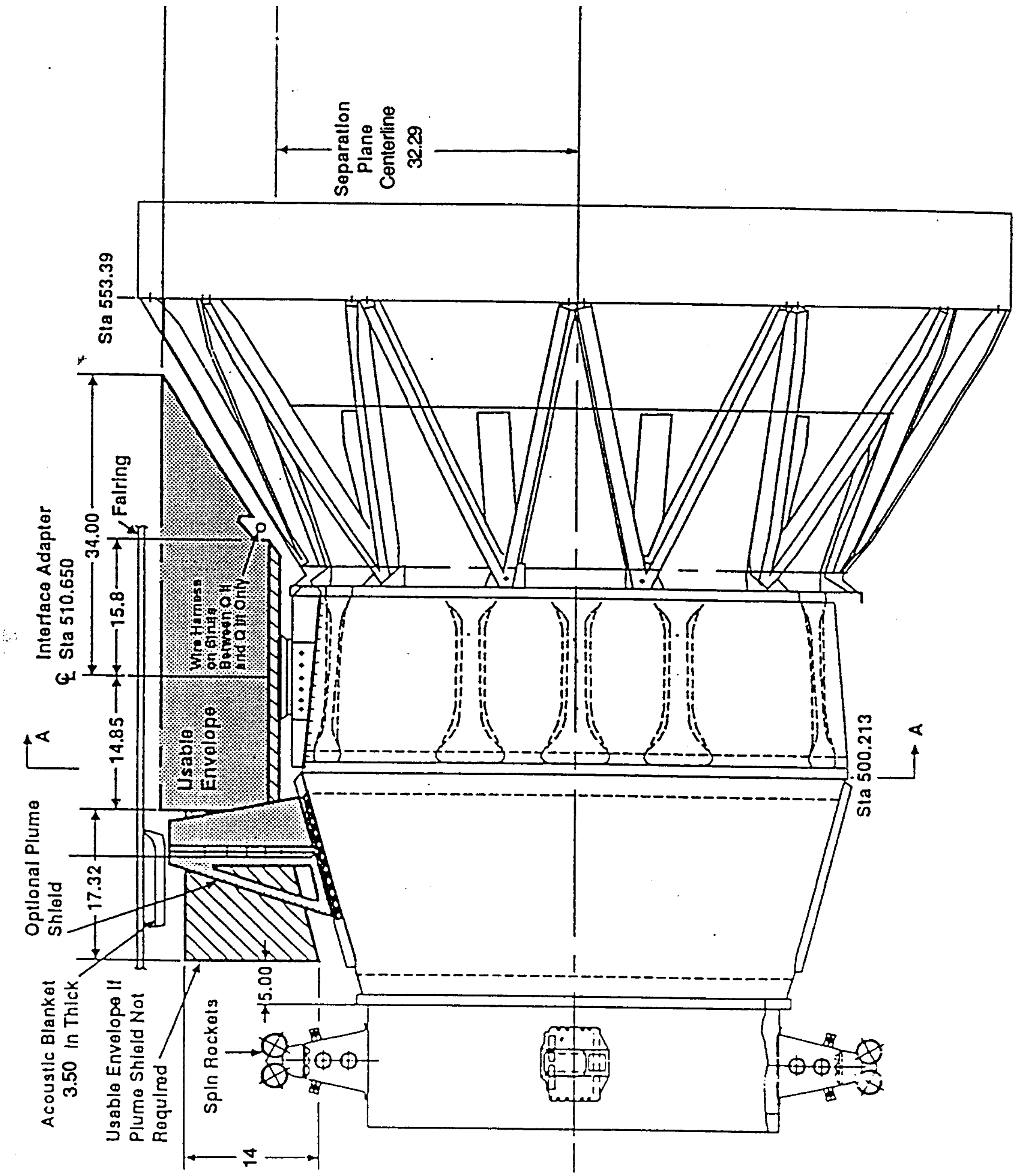


Figure 5.7.1.1a Usable Envelop for a Separating SP (SPE1), 9.5ft PLF Three-Stage Mission

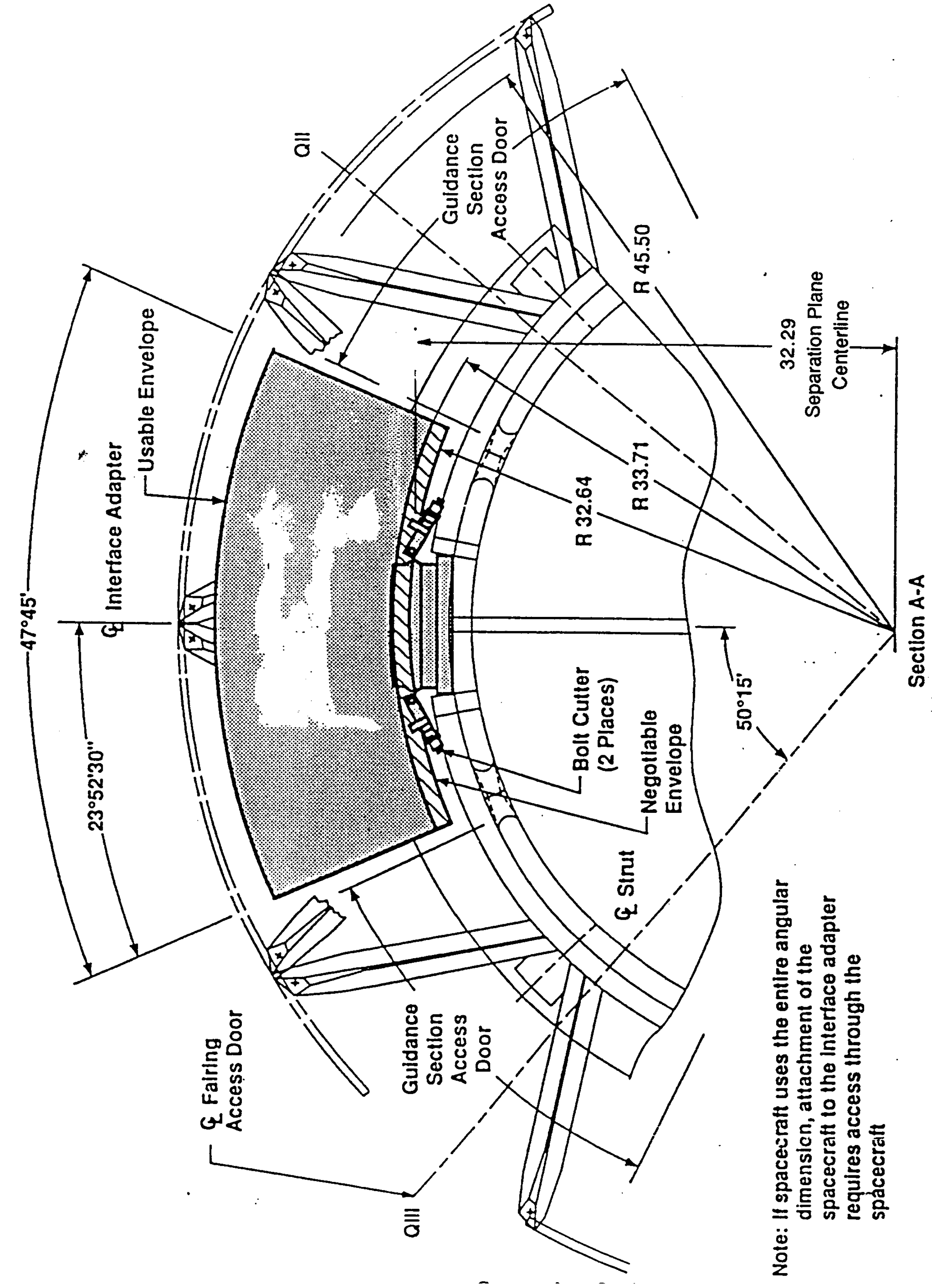


Figure 5.7.1.1b Usable Envelope for a Separating SP (SPE1), End View

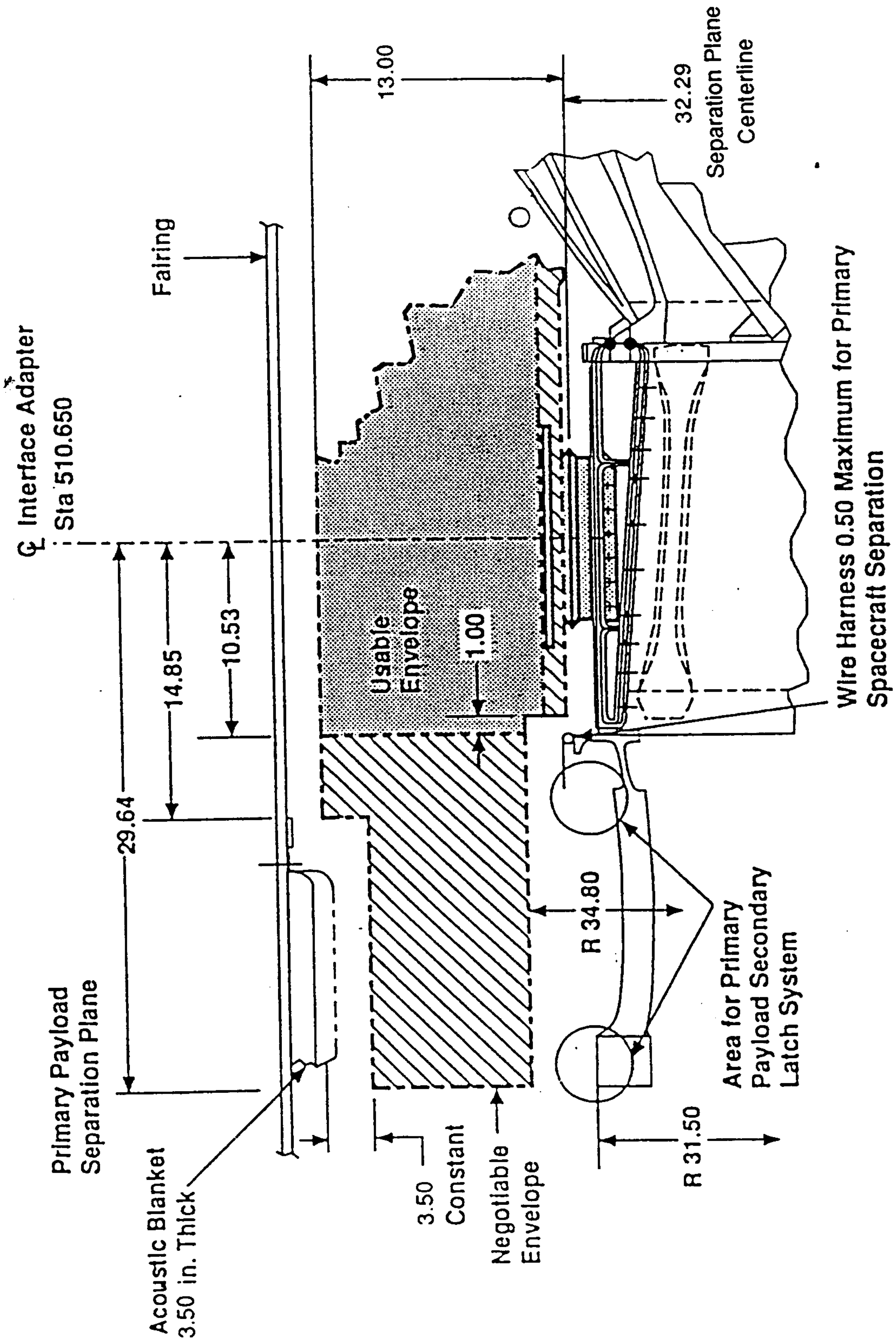


Figure 5.7.1.1c Usable Envelope for a Separating SP (SPE1), Two-Stage Mission, 6019 Primary Payload PAF

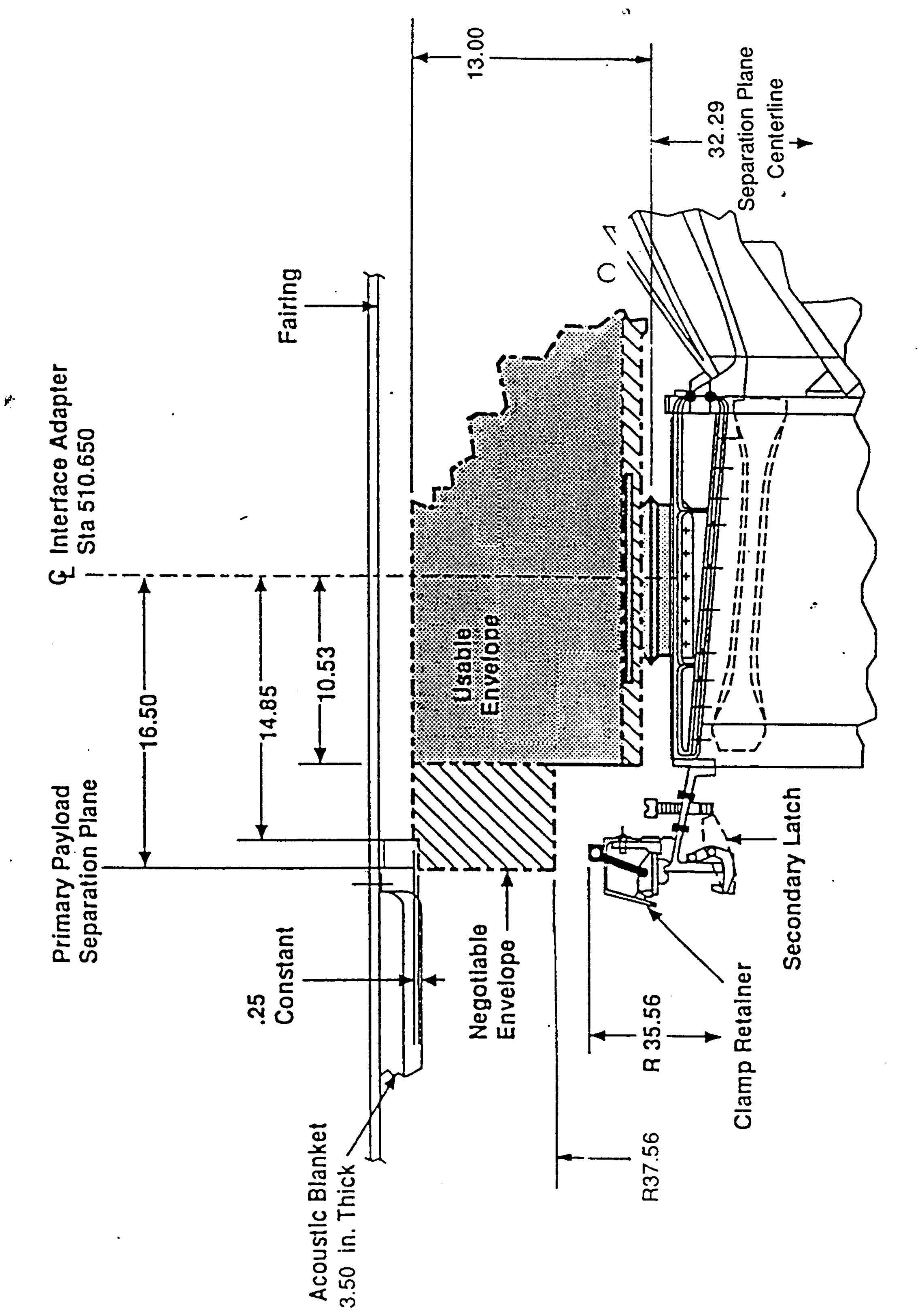


Figure 5.7.1.1d Usable Envelope for a Separating SP (SPE1), Two-Stage Mission, 6306 Primary PAF

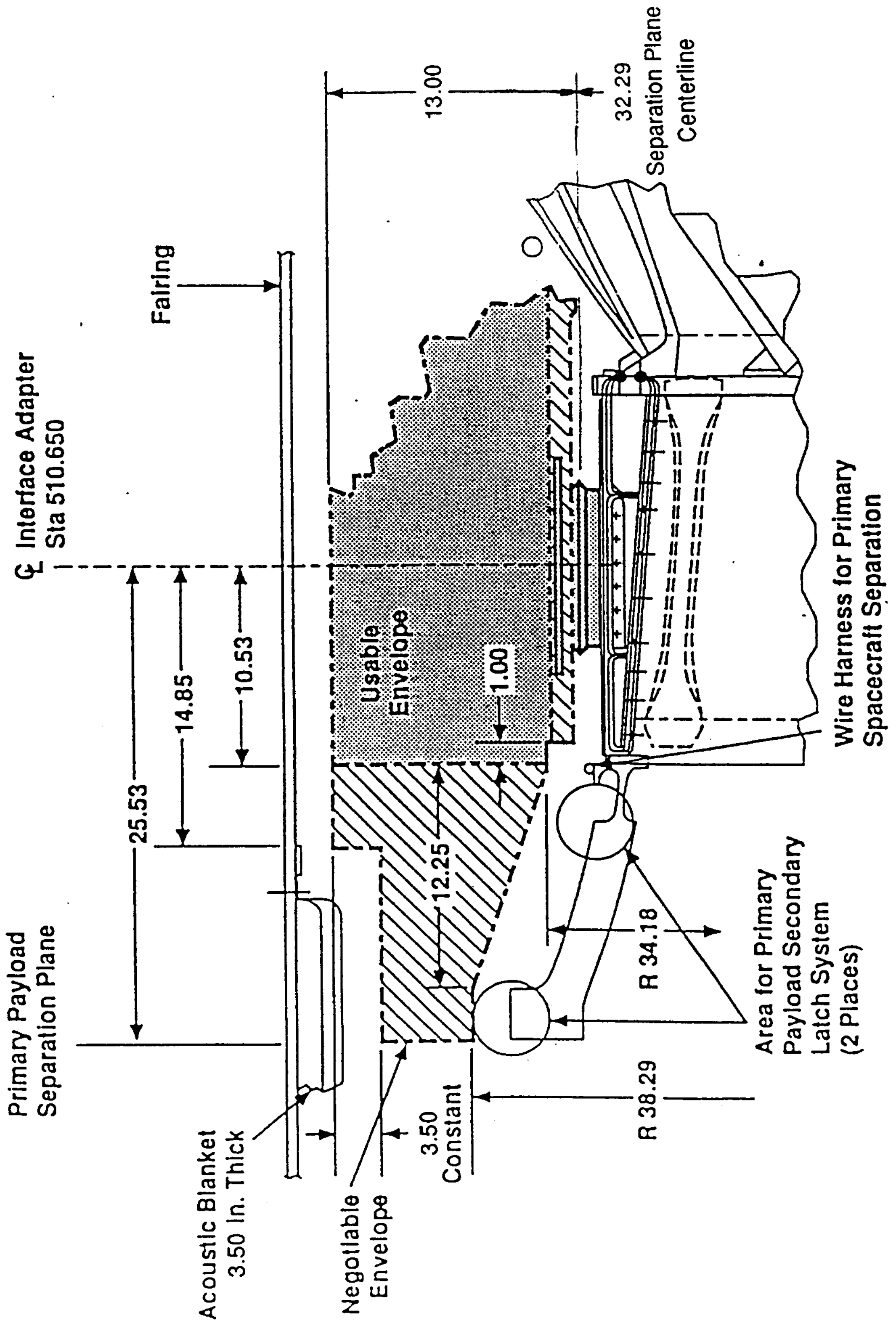


Figure 5.7.1.1e Usable Envelope for a Separating SP (SPE1), Two-Stage Mission, 6915 Primary Payload PAF

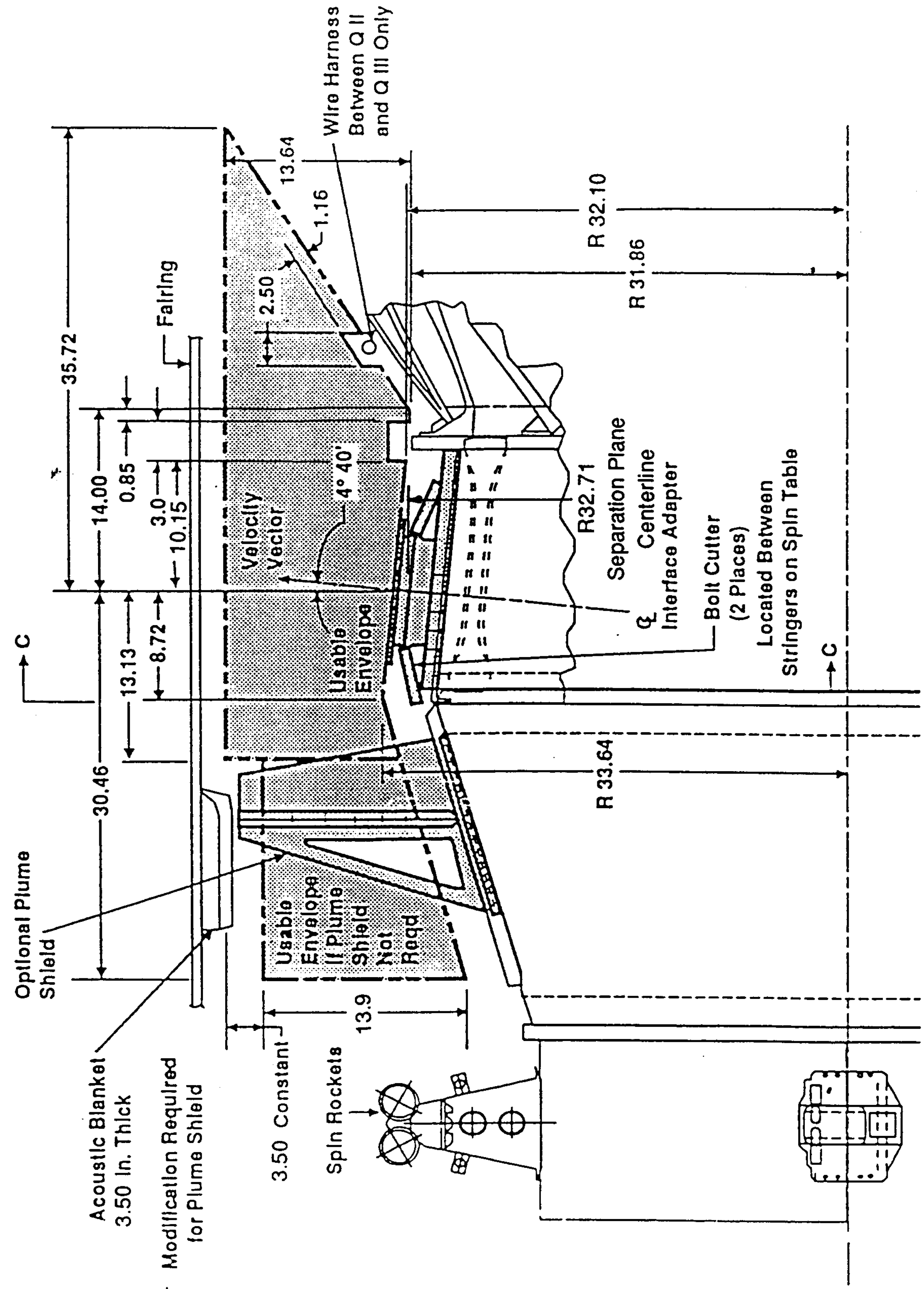


Figure 5.7.1.2 Usable Envelope for a Separating SP (SPE2), Three-Stage Mission

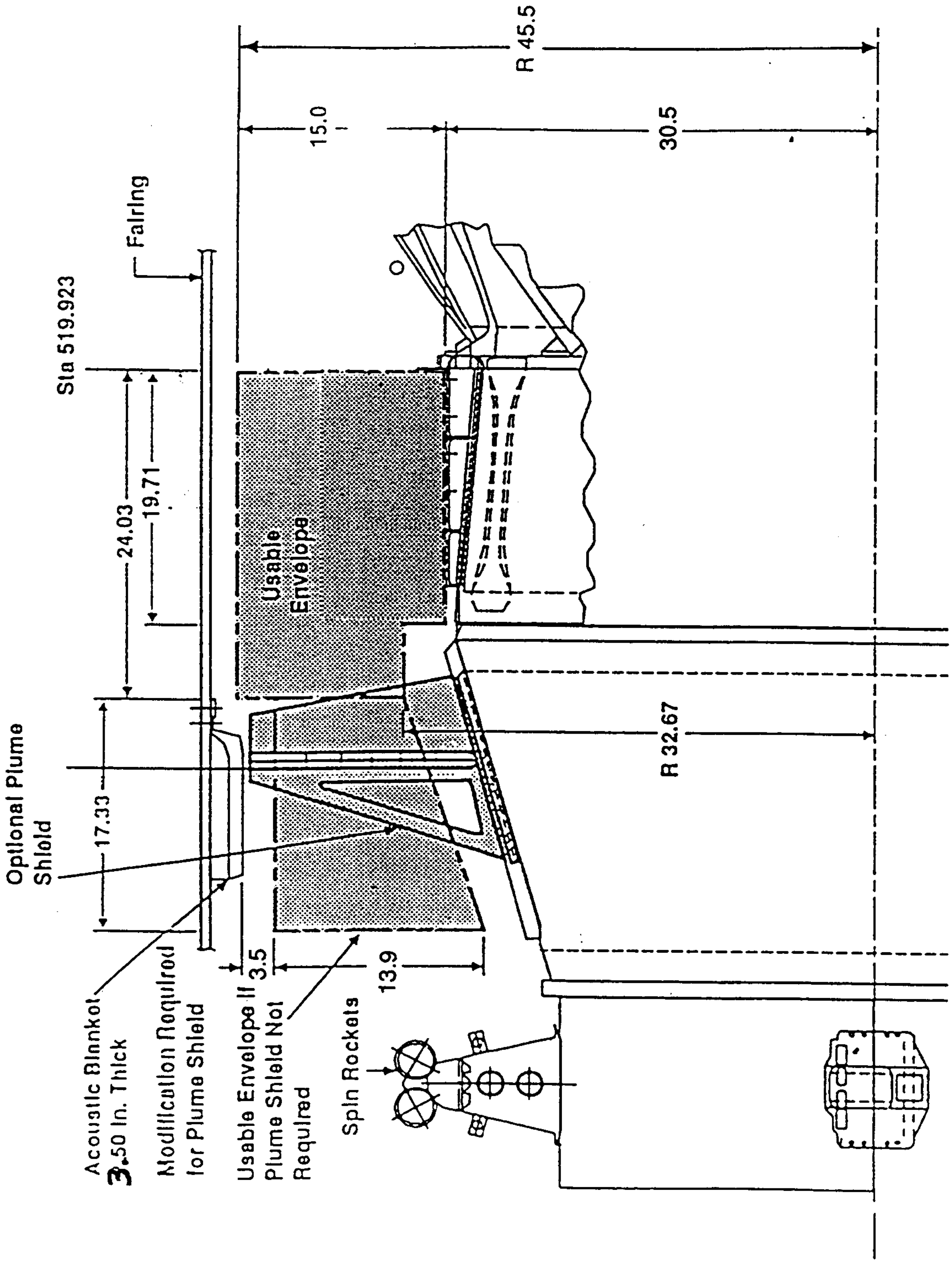


Figure 5.7.1.3a Example of a Usable Envelope for a Non-Separating SP, Three-Stage Mission

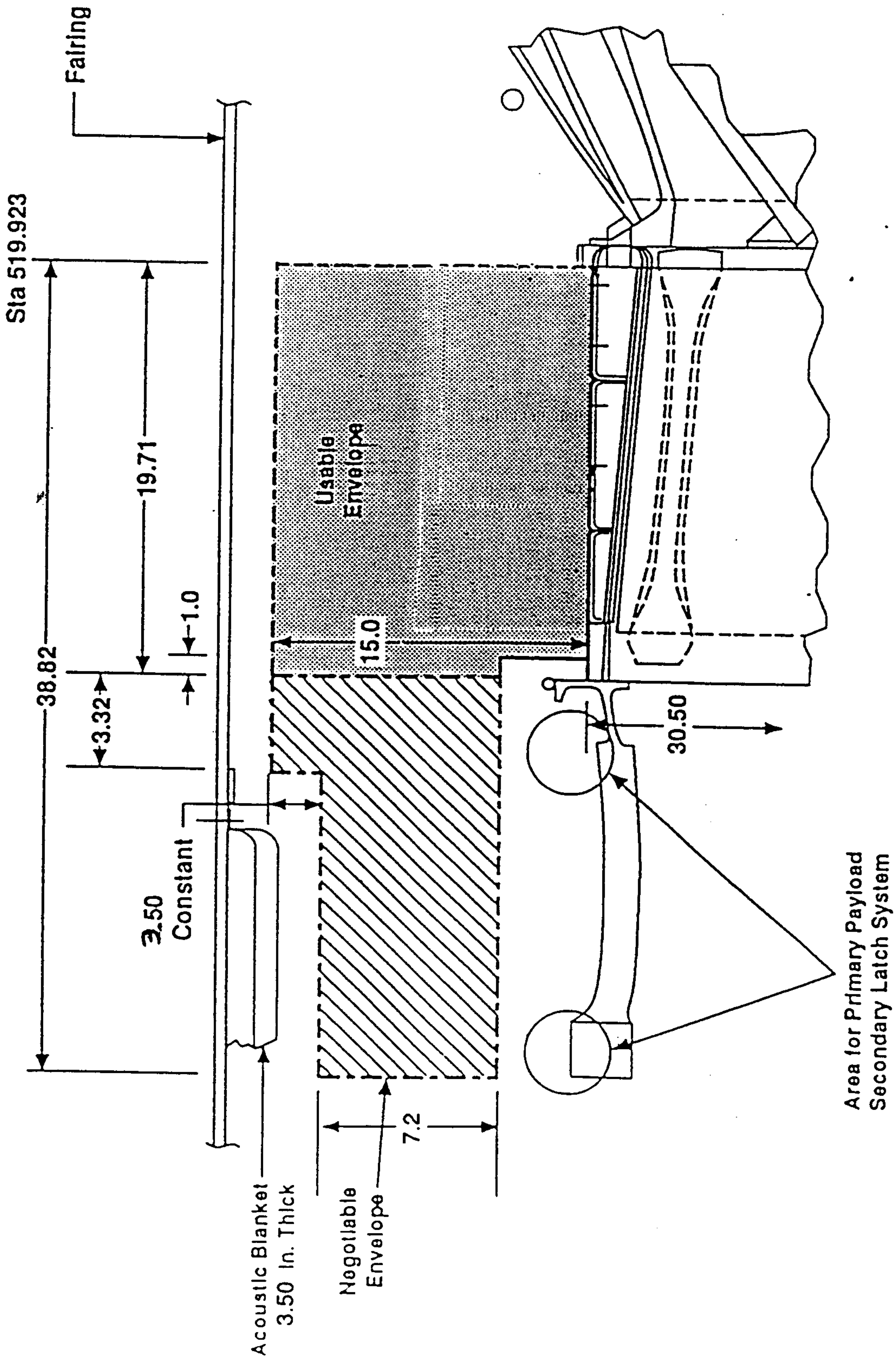


Figure 5.7.1.3b Example of a Usable Envelope for a Non-Separating SP, Two-Stage Mission, 6019 Primary Payload PAF.

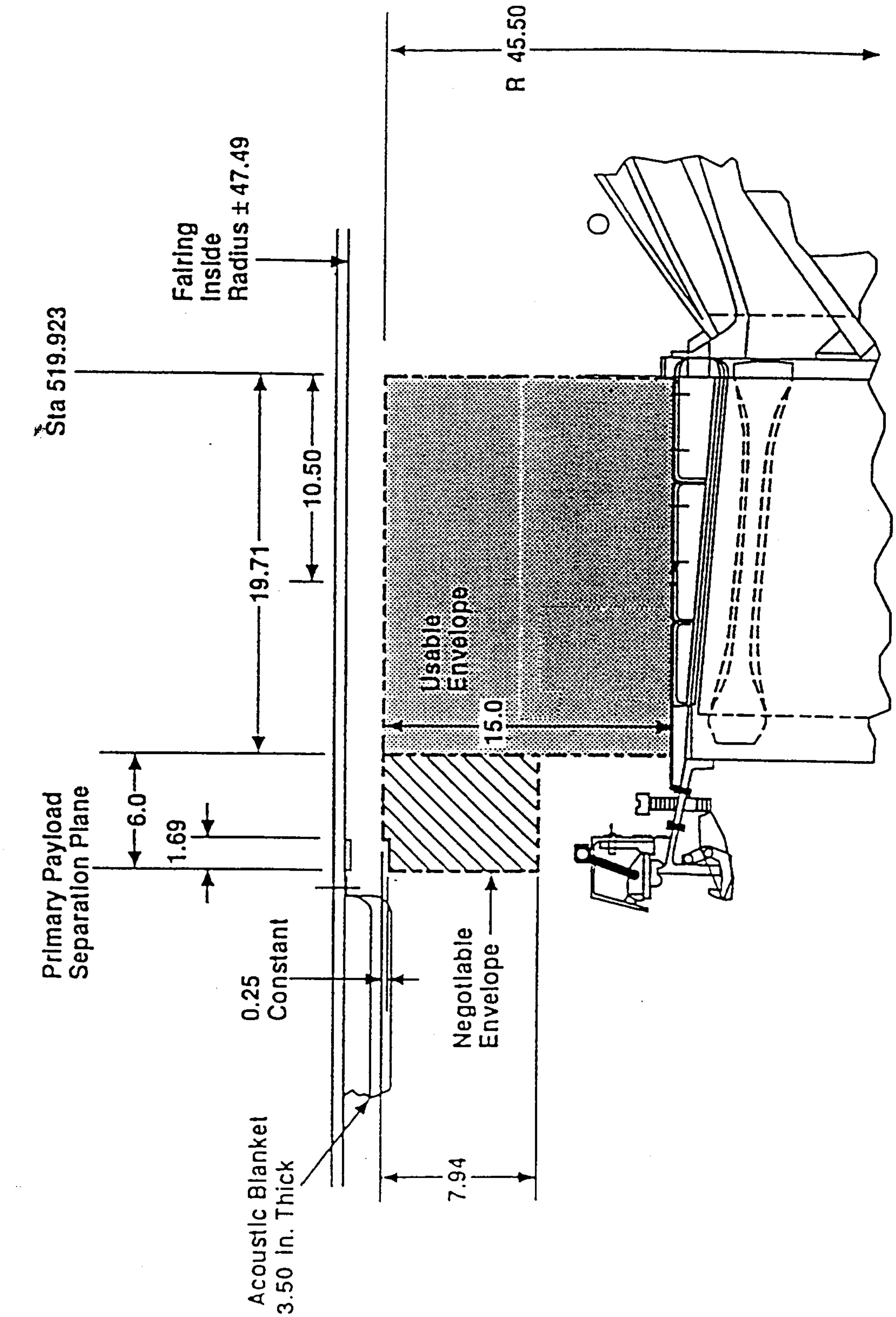


Figure 5.7.1.3c Example of a Usable Envelope for a Non-Separating SP, Two-Stage Mission, 6306 Primary Payload PAF.

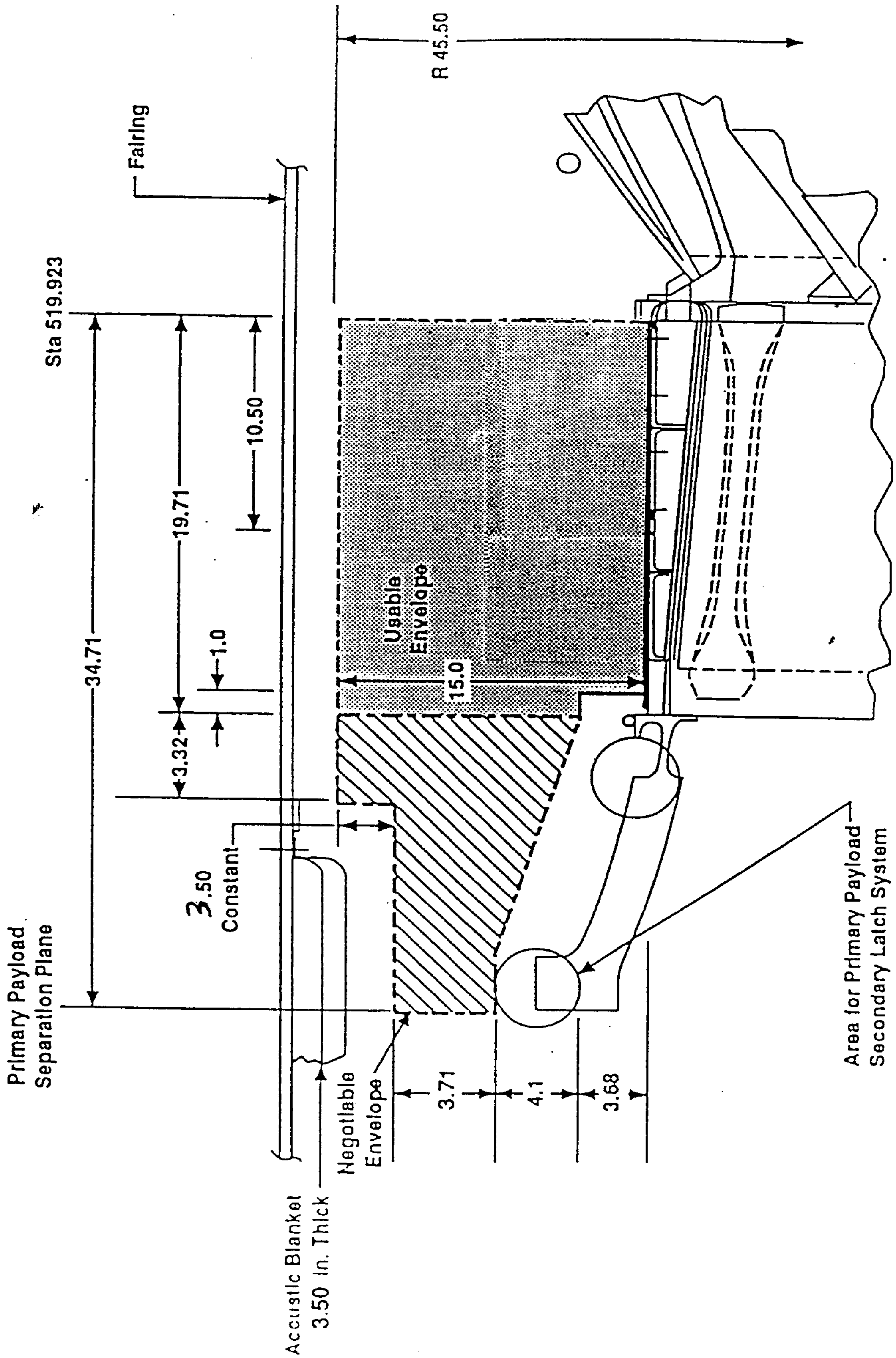


Figure 5.7.1.3d Example of a Usable Envelope for a Non-Separating SP, Two-Stage Mission, 6915 Primary Payload PAF.

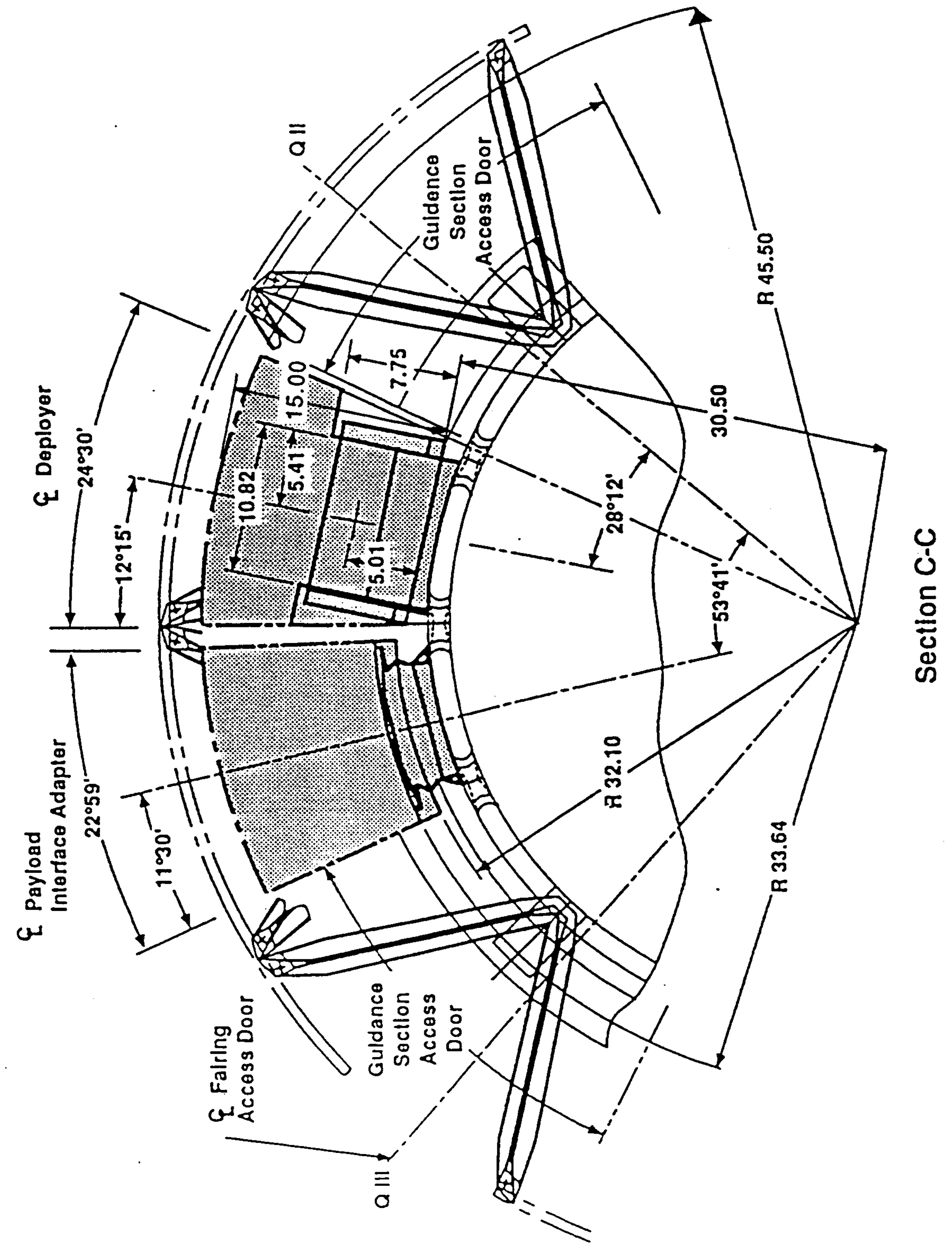


Figure 5.7.1.3e Example of a Usable Envelope, End View of Separating and Non-Separating SPs Side by Side

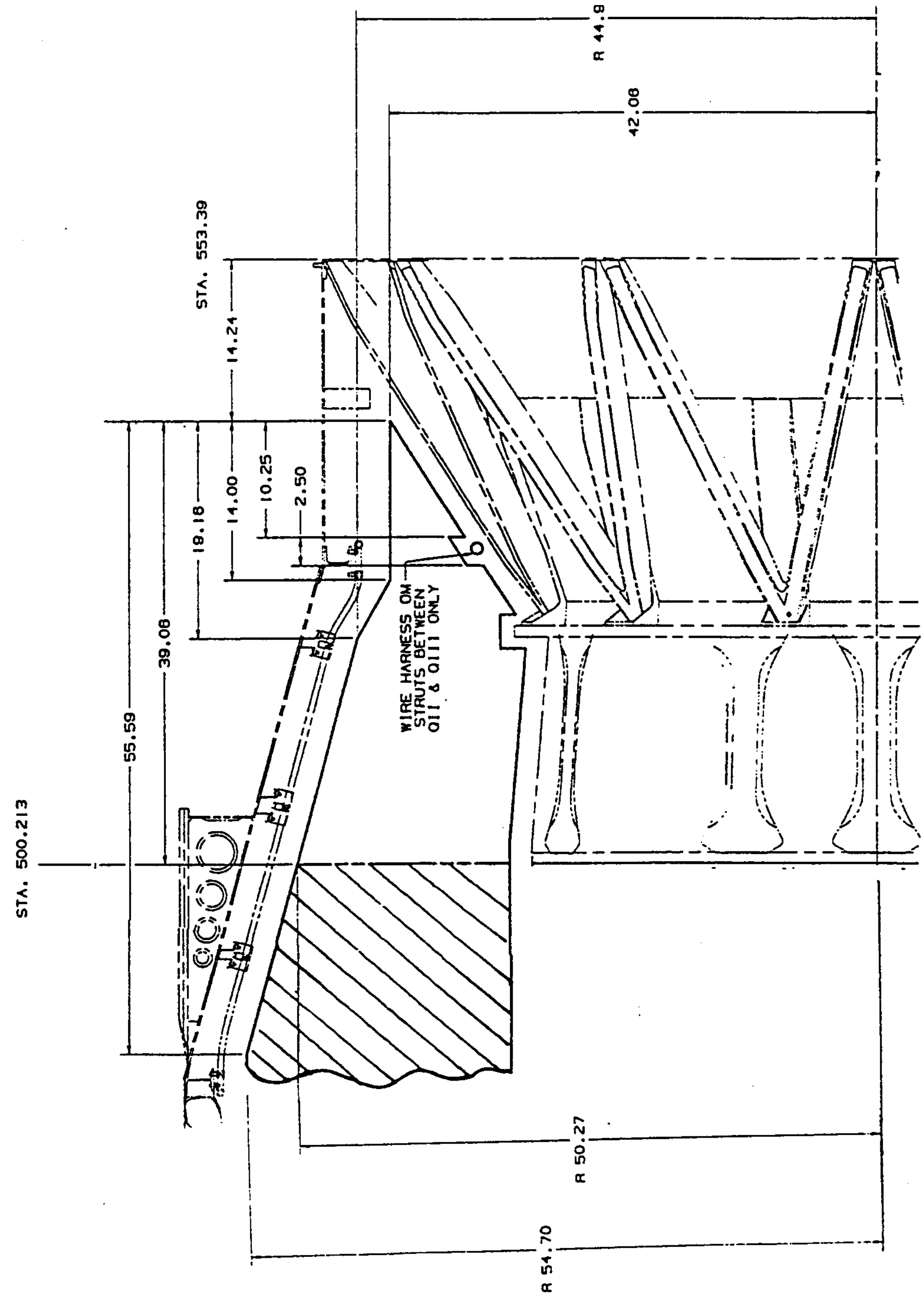


Figure 5.72 Usable Envelope for a Two-Stage Mission, 10-Foot Fairing, Separating SP

**Maximum Allowable Distance of Secondary Payload C.G.
Above Clampband Separation Plan (in)**

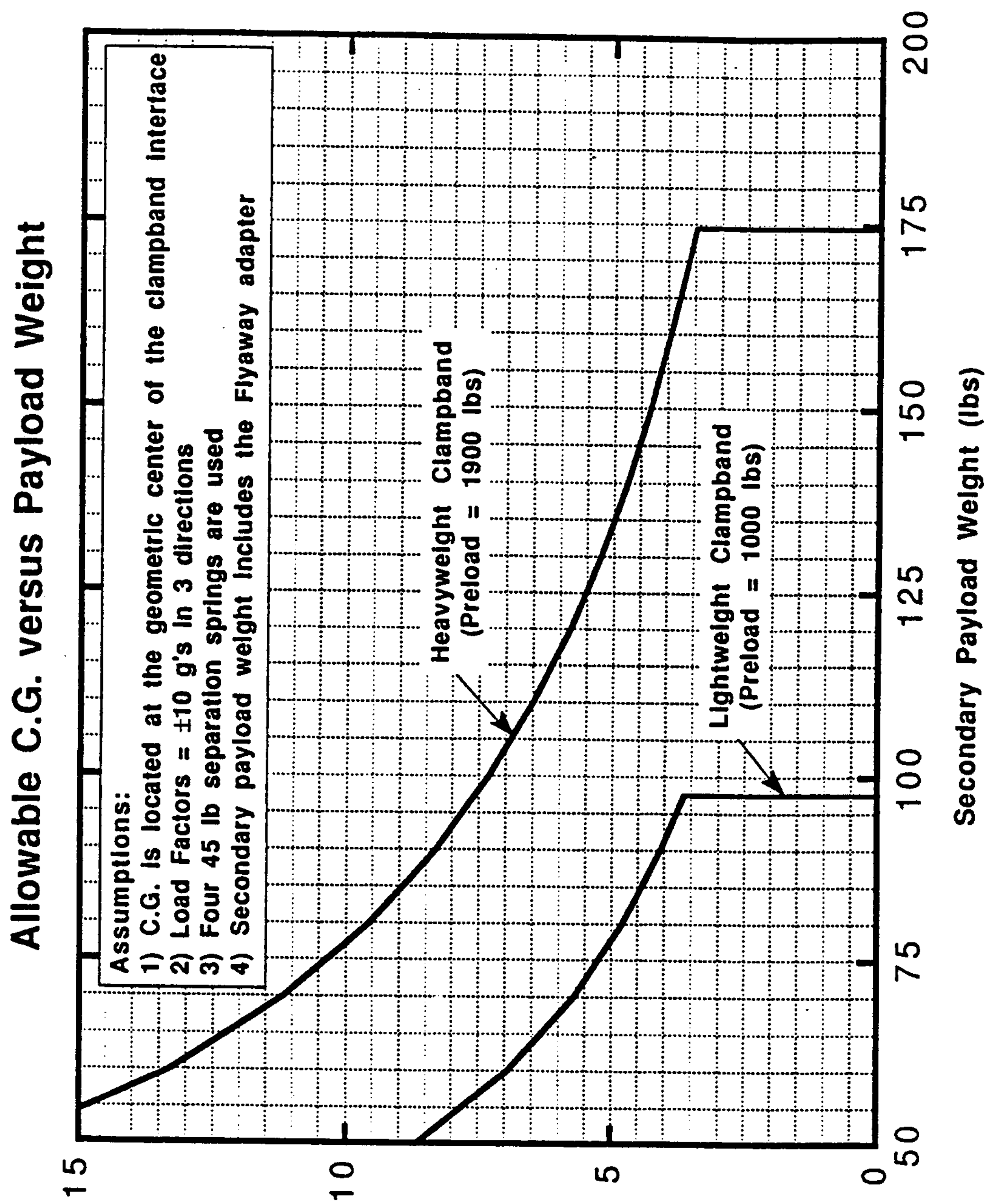


Figure 5.8 Structural Capability for Separating Payloads

5.9 ELECTRICAL INTERFACE

Presented in the following paragraphs is a description of the spacecraft/vehicle electrical interface design constraints.

5.9.1 Pre-Launch Testing

SPs do not have access to the spacecraft-to-blockhouse wiring utilized to monitor the primary spacecraft after vehicle mating. The first verification of the SP electrical interface is performed in Delta Mission Checkout (DMCO) dual composite test approximately 10 to 15 weeks prior to launch. The second verification with the launch vehicle is on the pad during the F12 simulated flight about 3 weeks prior to launch. After mating with the vehicle, the electrical interface for monitoring or trickle charging is through SP-provided drag-on cables. Final verification of the electrical interface is performed during Flight Program Verification (F6). No SP electrical checkout should be planned after F6.

5.9.2 Flight support

In order to assure the primary payload of no interference from the SP, any vehicle electrical power or telemetry support required by the SP is typically separate from primary payload support systems. MDA supplies power for separation of the SP clamp band with a 1 amp-hour (HR-1) battery. An additional HR1 has been provided for SP power on certain missions. SPs should plan on providing their own power within the defined SP envelopes. However, if required, MDA may be able to provide power from an SP dedicated 1 or 5 amp-hour battery.

Limited capability to provide discrete commands from the flight computer may be available from the second stage.

In the current Delta system, SP telemetry data may be passed either through a single separate low-rate channel of the second stage system or through a separate system provided by the SP. For missions flown in 1995 and beyond, a new Delta avionics system is planned. Although the new system will not support SP telemetry, modifications will be made to support a single channel of low rate data. Alternatively, the SP may supply an approved system.

Section 6
Launch Site Activity

The Delta II expendable launch vehicle (ELV) is launched from the Eastern Range (ER) at Cape Canaveral Air Force Station (CCAFS) or the Western Range (WR) at VAFB.

6.1 SPACECRAFT PROCESSING

MDA is the Delta launch system operator and maintains a launch team which provides launch services to commercial users, NASA, and the USAF at CCAFS and VAFB. NASA at Kennedy Space Center (KSC) designates a Launch Site Support Manager who arranges all the support requested from NASA for a launch from CCAFS. These services include range instrumentation, facilities/equipment operation and maintenance, as well as safety, security, and logistics support. Requirements satisfied by NASA and/or USAF are described in the government's universal documentation system format. MDA and the spacecraft agency generate the Program Requirements Documents (PRD). Formal submittal of these documents to the government agencies are made by NASA.

In addition to those facilities required for the Delta II launch vehicle, specialized facilities are provided for checkout and preparation of the spacecraft at each range. Laboratories, clean rooms, receiving and shipping areas, hazardous operations areas, offices, etc., are provided at CCAFS and VAFB for use by spacecraft project personnel.

The commonly used launch site spacecraft facilities are as follows:

ER spacecraft payload processing facilities:

NASA-provided hangars: AO, AM, AE, or S in the CCAFS industrial area.

ER hazardous processing facilities (HPFs):

NASA-provided explosive safe area 60, SAEF 2, CHSF at KSC, or CCAFS.

WR spacecraft payload processing facilities:

NASA-provided hangars: Building 836.

WR hazardous processing facilities (HPFs):

Building 1610

The spacecraft agency must provide its own test equipment for spacecraft preparations.

Transportation and handling is provided by the SP agency for the spacecraft and associated equipment from any of the local airports to the spacecraft processing facilities. Equipment and personnel are available for loading and unloading operations. A handling container, if compatible, with the SP design, is available for transportation of the SP from the processing facility to the launch site. Its internal dimensions are 139.7 cm (55.0 in) by 107.9 cm (42.5 in) with a usable height of about 39.6 cm (15.6 in). The spacecraft project may elect to use its own handling container. If the latter option is selected by the spacecraft project, compatibility of the spacecraft container and the hoisting and handling systems on the MST must be determined. KSC will provide transportation from the spacecraft processing facilities to the MST, and MDA will hoist necessary equipment into the MST.

Shipping and handling of hazardous materials such as EEDS, radioactive sources, etc., must be in accordance with applicable regulations. It is the responsibility of the spacecraft agency to identify these items and become familiar with such regulations. These regulations include those imposed by NASA, USAF, and DOT (refer to Section 9).

Typical activities for the last seven days before a launch are shown in Figures 6-la through 6.1d. These figures include operations for the SEDS-1 SP.

The extent of spacecraft field testing varies and is determined by the spacecraft agency. The spacecraft launch vehicle schedule is integrated and is similar from mission to mission from the time the spacecraft is transported to the launch site.

Non-separating SPS should be available for integration approximately 10 days prior to launch. Separating SPs require an additional 3-5 days for clamp band installation.

The following is a typical schedule of integrated activities shown in flight minus (F-) days. Saturdays, Sundays, and holidays are not scheduled work days and therefore, are not F- days. The F- days, from spacecraft mate through launch, are coordinated with each spacecraft agency to optimize testing. SP operations must be worked around conflicting primary spacecraft operations. All operations are formally conducted and controlled using launch preparation documents. The schedule of spacecraft activities during that time is controlled by the MDA Launch Operations Manager.

F-7 Tasks include transportation to the launch site, erection, and mating of the primary spacecraft to the Delta II second stage in the MST cleanroom. Preparations are made for the launch vehicle flight program verification test (Figure 6.1b). Mating of an SP is scheduled after mating of the primary spacecraft. However, to avoid exposing the primary spacecraft to an open MST, SPs which are hoisted into the white room must arrive at the launch site I day before mating of the primary spacecraft. Due to primary spacecraft schedules, some missions may require that the SP be installed in the second shift, between F7 and F6. On-pad operations and checkout of an SP should be kept to a minimum.

F-6 Tasks include the launch vehicle flight program verification test, followed by the vehicle power-on stray voltage test. Spacecraft systems to be powered at liftoff are turned on during the flight program verification test and all data is monitored for electromagnetic interference (EMI) and radio frequency interference (RFI). Spacecraft systems to be turned on at any time between F-5 day and spacecraft separation are turned on for approximately 20 minutes in support of the vehicle power-on stray voltage test. Spacecraft support of these two vehicle system tests is critical to meet the scheduled launch date (Figure 6.1b). Participation by the SP project in this test is the final opportunity for electrical checkout of the SP prior to launch.

F-5 Tasks include Delta II vehicle ordnance installation and connection, and preparations for fairing installation (Figure 6.1c).

F-4 Tasks include spacecraft final preparations prior to fairing installation, including Delta II upper-stage closeout, and fairing installation (Figure 6.1c). SP closeout should be completed prior to fairing installation. Typically, only protective cover removal is allowed after fairing installation.

F-3 Tasks include preparations for second-stage propellant servicing, and fairing finaling.

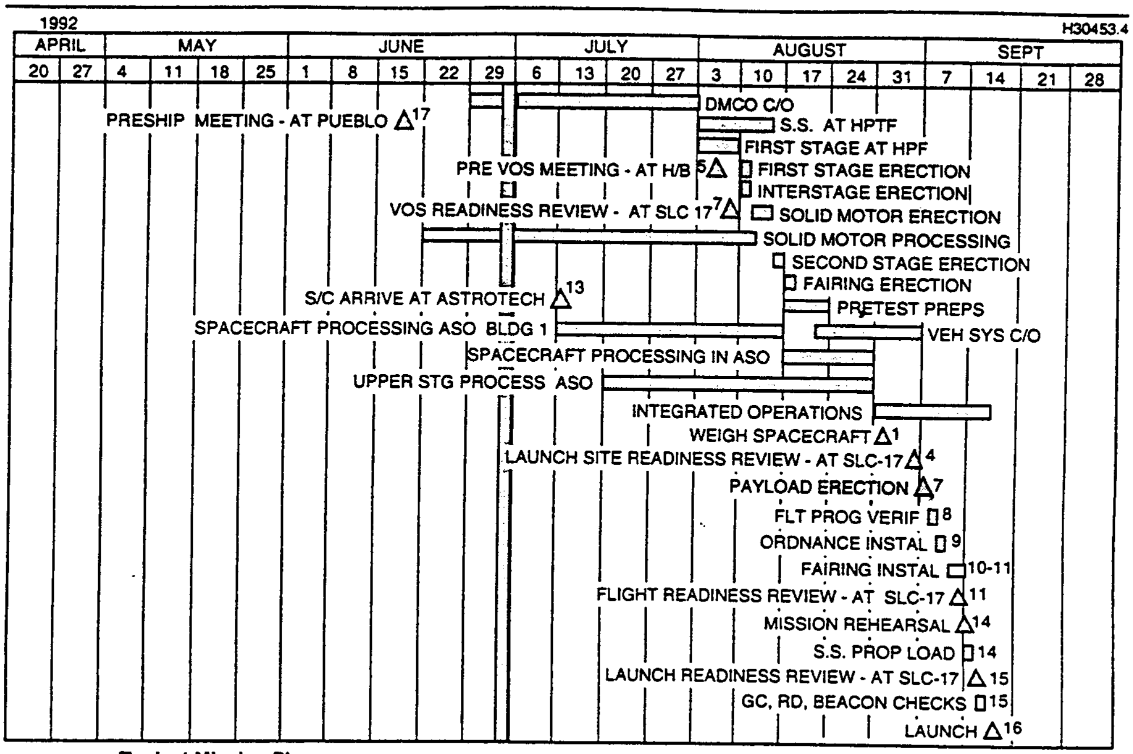
F-2 Tasks include propellant loading of the second stage.

F-I Tasks include launch vehicle guidance system turn-on; C-band beacon readout; and guidance system azimuth update, followed by the vehicle Class A ordnance connection, spacecraft ordnance arming, and final fairing preparations for MST removal, second-stage engine section closeout, and launch vehicle final preparations.

F-0 Launch day preparations include gantry removal, final arming, and terminal sequences and launch. The nominal hold and recycle point is T-4 minutes (Figure 6.1d).

Typical schedules for pre-launch preparations at SLC-17, preparation of the Delta II upper stage in the HPF, and the last seven days of the countdown are provided in Figure 6.1. These schedules are integrated and tailored to meet the needs of the primary spacecraft project. As noted above, SPs are generally mated to the second stage just after primary spacecraft mating.

The SP may be transported and erected into the white room by two methods. If it is small and light enough to be hand carried, the elevator and stairs can be used on or before the day of primary spacecraft erection (F7). Otherwise the SP will be hoisted into the white room via the MST crane at least one day prior to primary payload erection. The OLS provided shipping container requires the use of the crane (see figures 6.le and f). SP provided transport containers must meet specific safety and material requirements. Details will be provided by MDA.



Typical Mission Plan

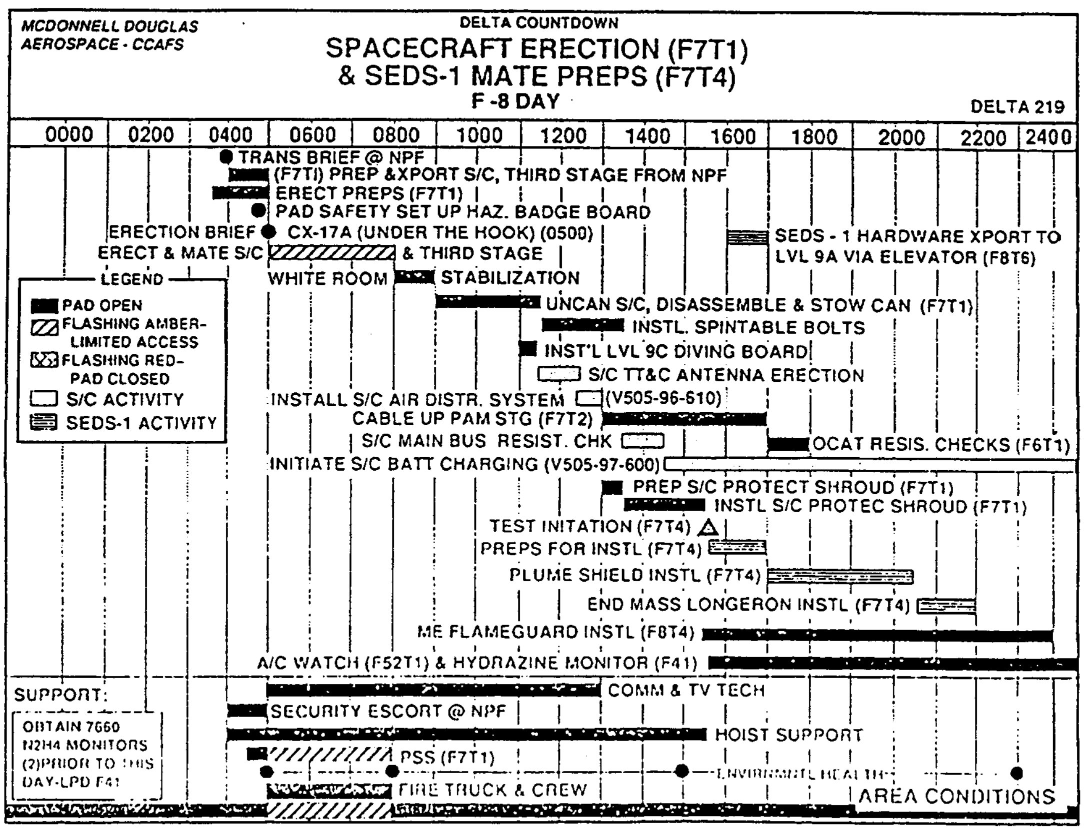
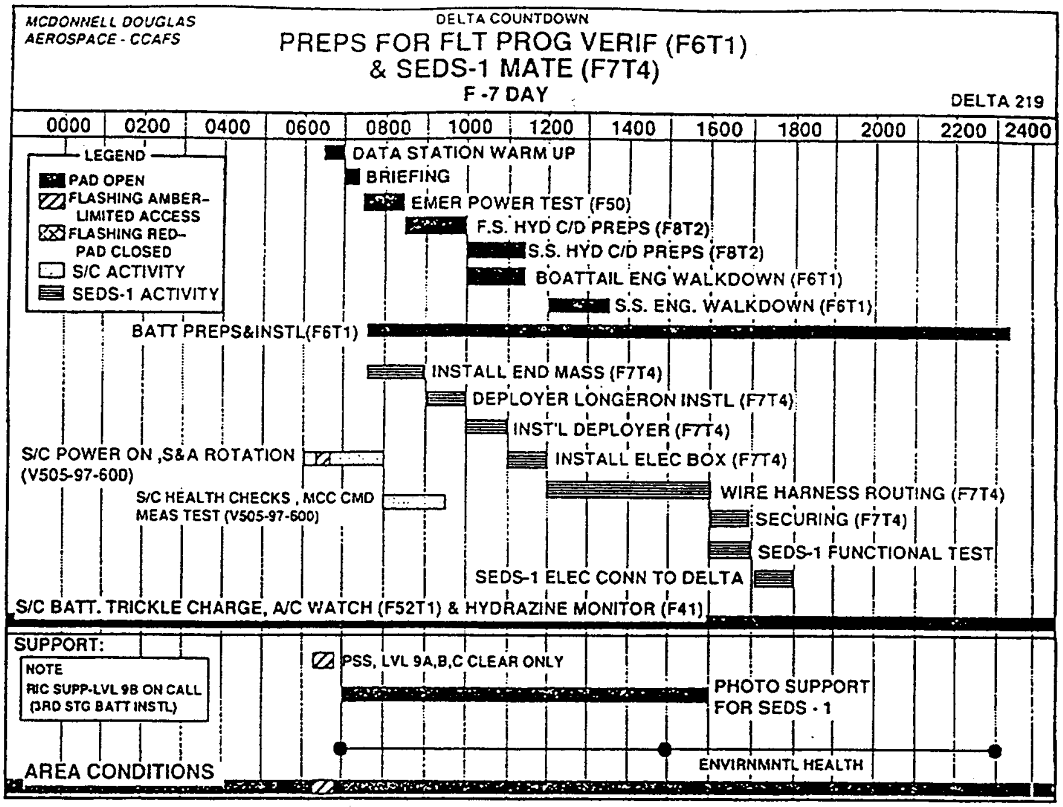


Figure 6.1a Launch Preparation Schedule



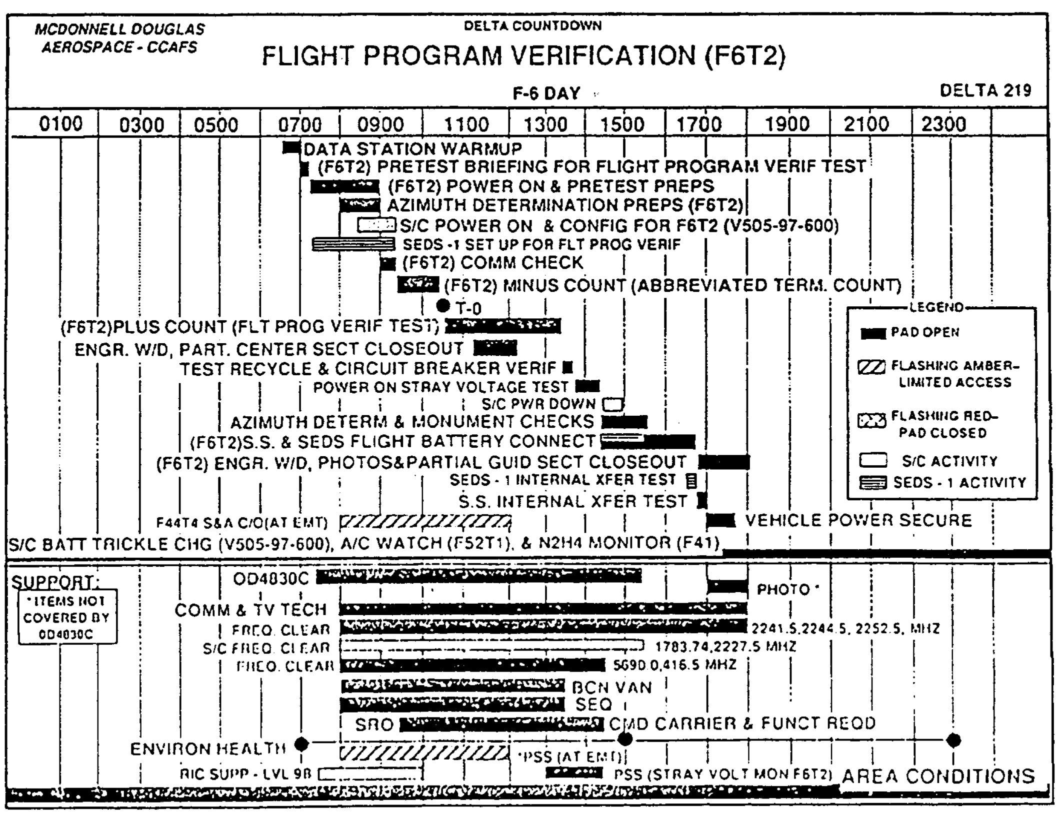
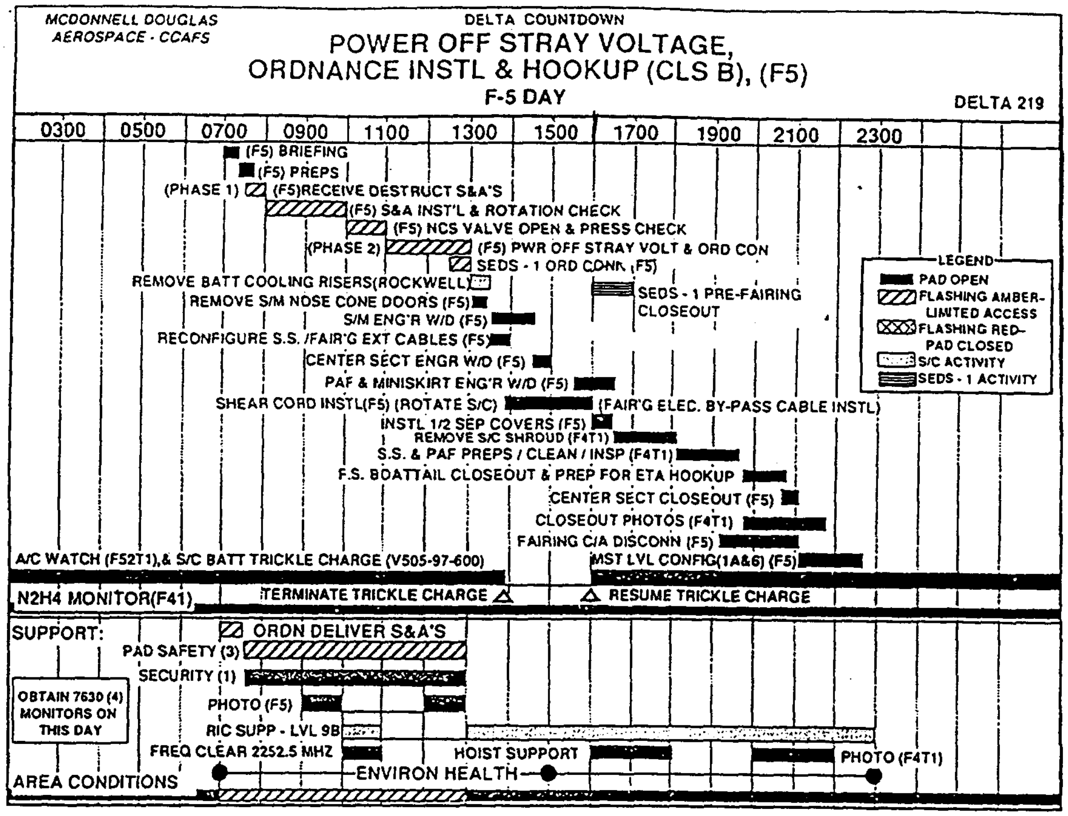


Figure 6.1b Launch Preparation Schedule



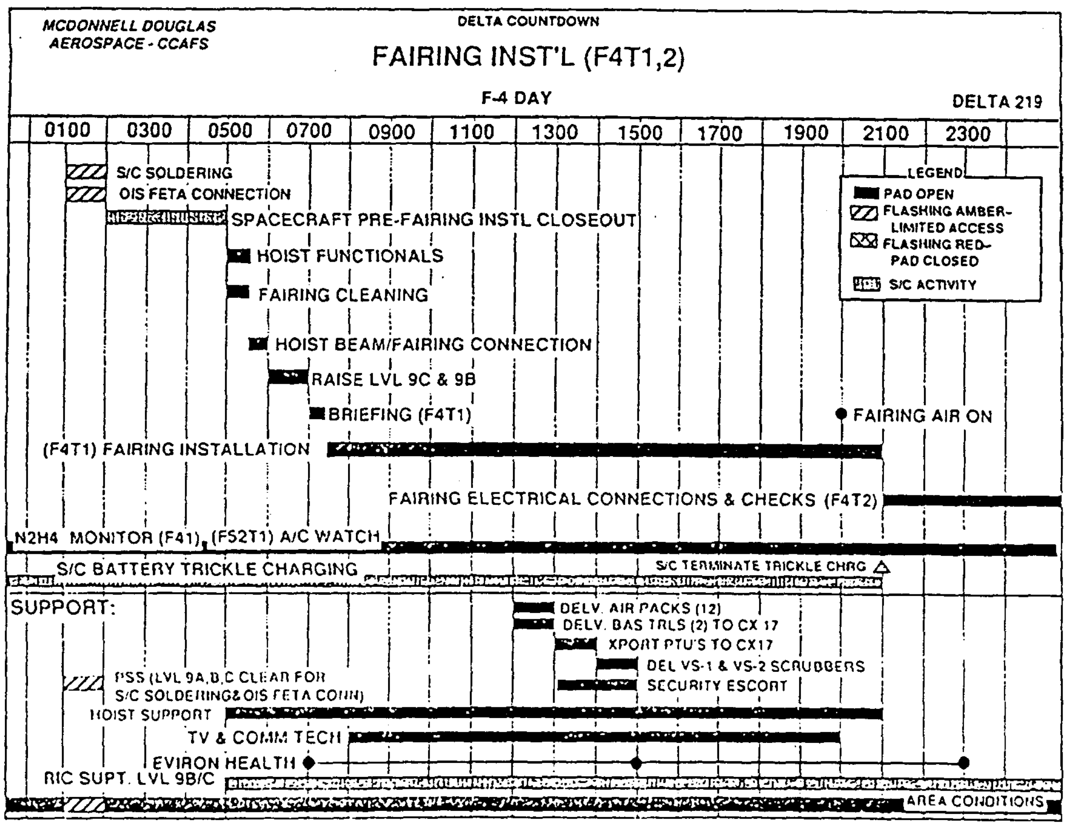
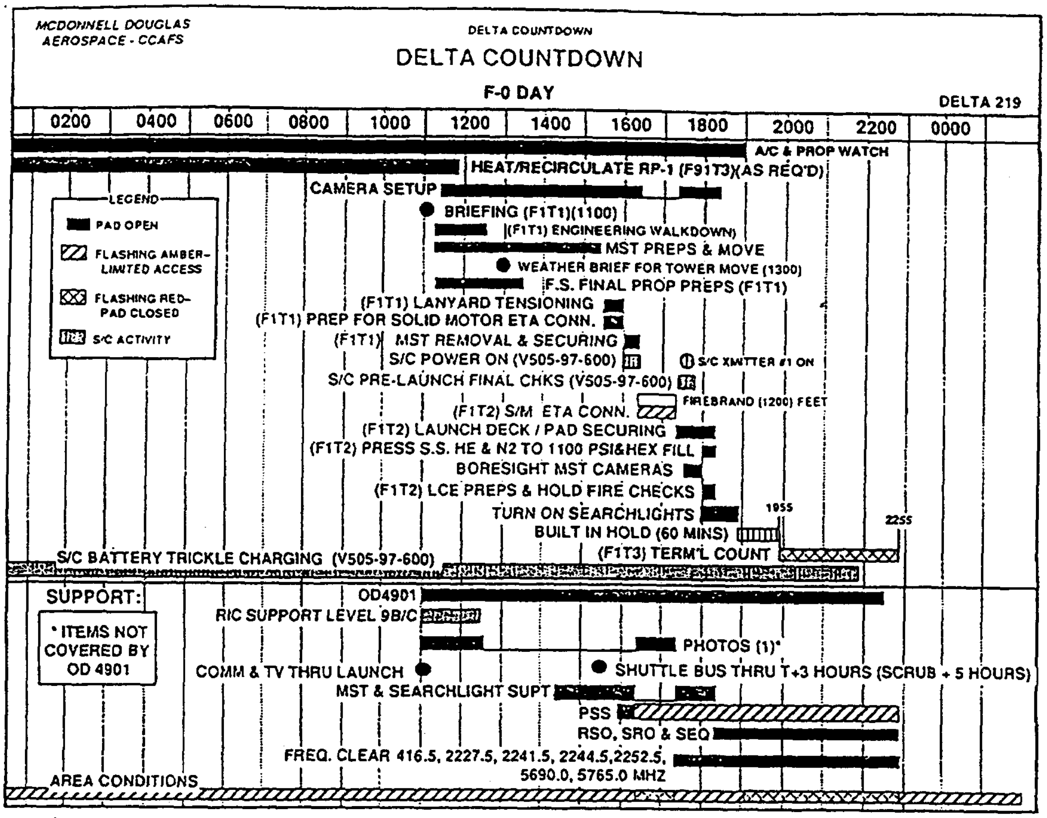


Figure 6.1c Launch Preparation Schedule



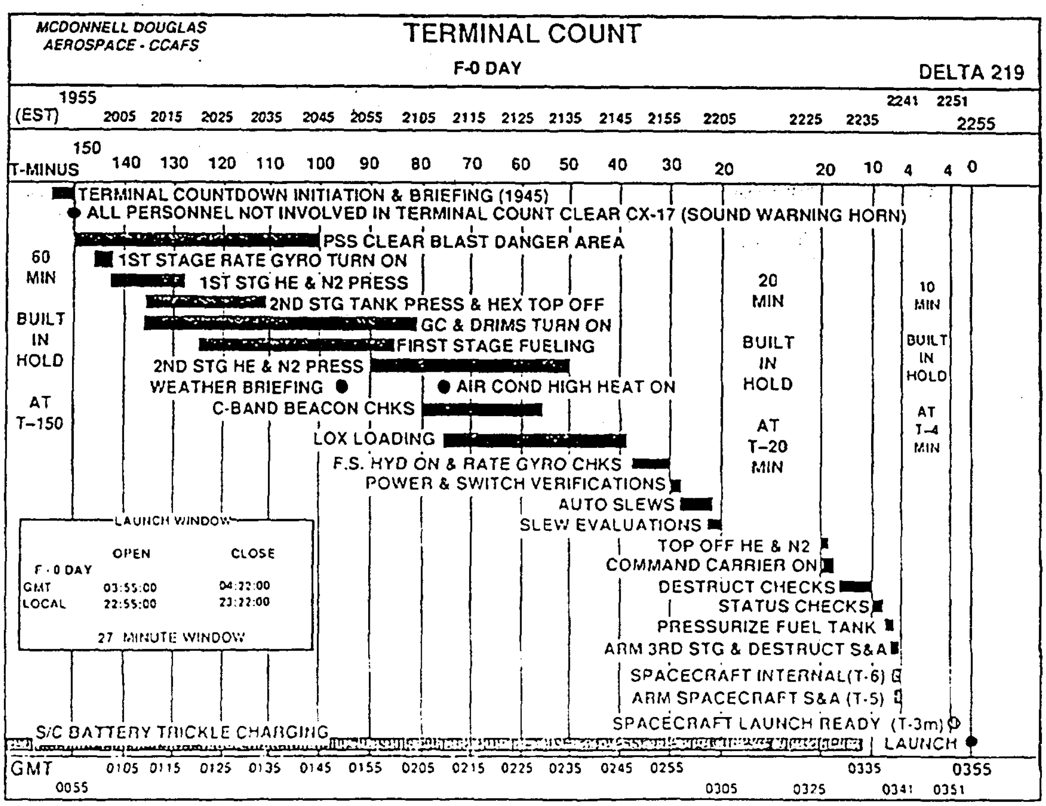


Figure 6.1d Launch Preparation Schedule

Installation of the SP is typically performed using an MDA supplied GSE adapter and the "iron maiden" installation fixture (see figures 6.lg and h). The adapter adjusts for various SP sizes.

6.2 LAUNCH SITE

SLC-17 is located in the southeastern section of CCAFS. It consists of two launch pads (17A and 17B), a blockhouse, ready room, shops, and other "facilities needed to prepare, service, and launch the Delta II vehicle. The VAFB launch site has a single pad, SLC-2W.

Since all operations in the launch complex area involve or are conducted in the vicinity of liquid or solid propellants and explosive ordnance devices, strict regulations are in effect with regard to the number of personnel permitted in the area, safety clothing to be worn, the type of activity permitted, and equipment allowed. Adherence to all safety regulations specified in Section 9 is required. Safety briefings on these subjects are given for those required to work in the launch complex area.

6.2.1 MST Spacecraft Work Levels

The number of personnel admitted to the MST is governed by safety requirements and by the limited amount of work space on the spacecraft levels. The relationship of the vehicle to the MST is shown in Figure 6.2.1a. Floor plans for both CCAFS and VAFB are shown in Figure 6.2.1b. Outlets for breathing air, helium, and nitrogen are provided on MST levels 9A and 9B.

Communications equipment provided on MST levels 9A and 9B includes telephones and operational communications stations for testing communications.

6.2.2 Blockhouse

Launch operations are controlled from the blockhouse which is equipped with vehicle monitoring and control equipment. Space and a console are allocated for use by primary spacecraft personnel.

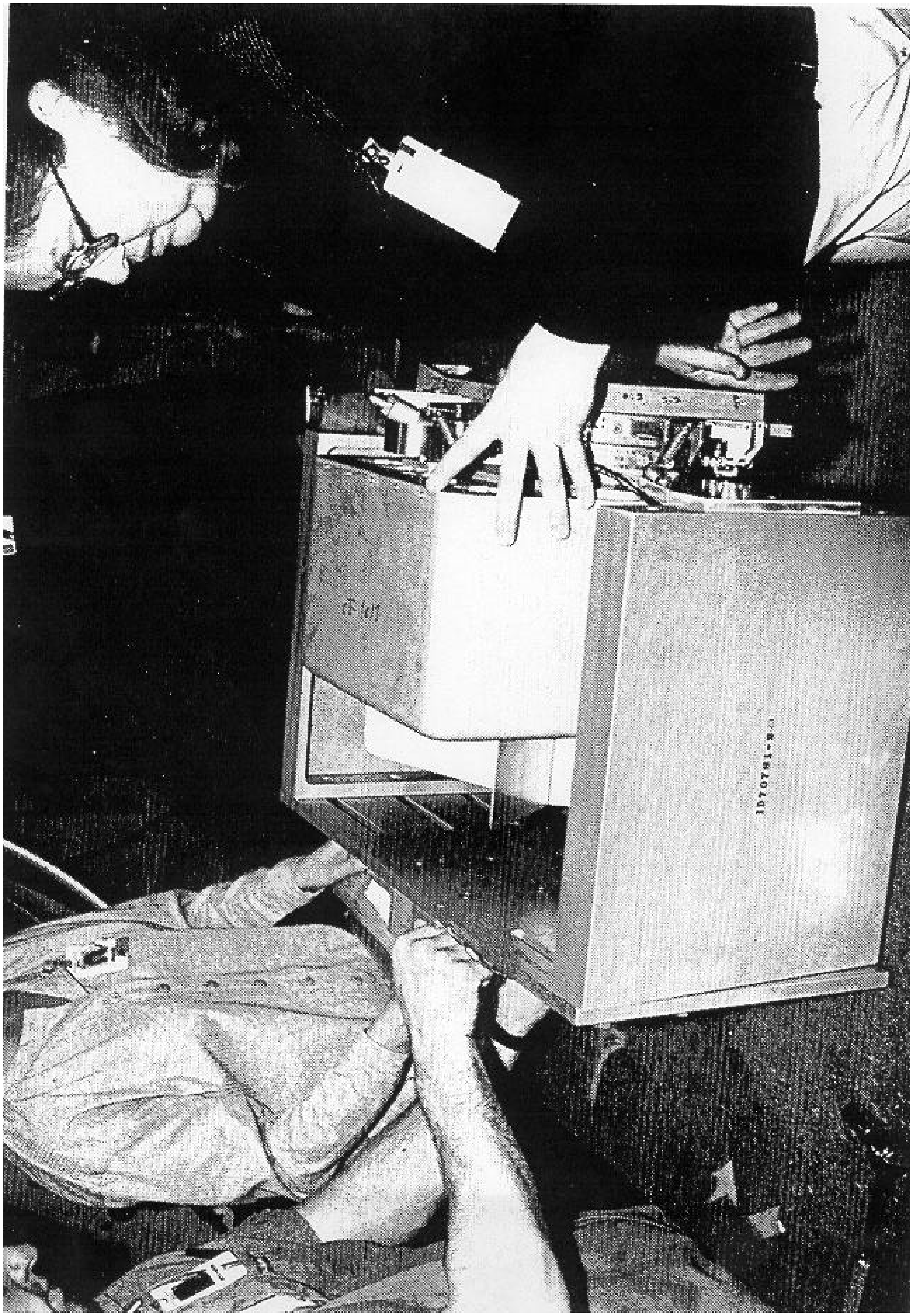
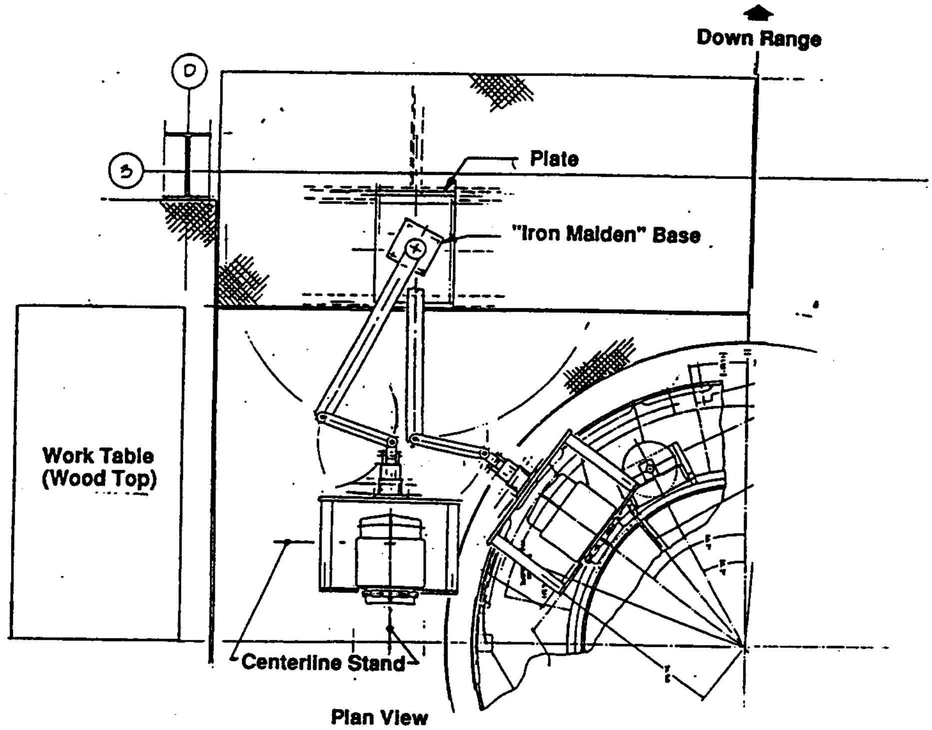


Figure 6.1g GSE Adapter



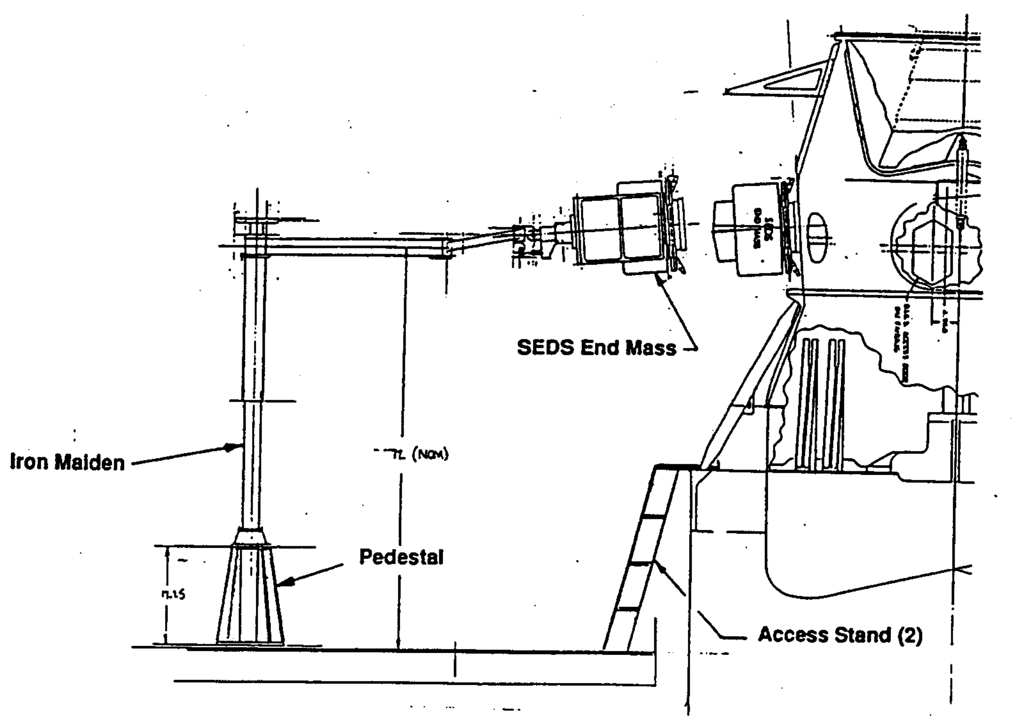
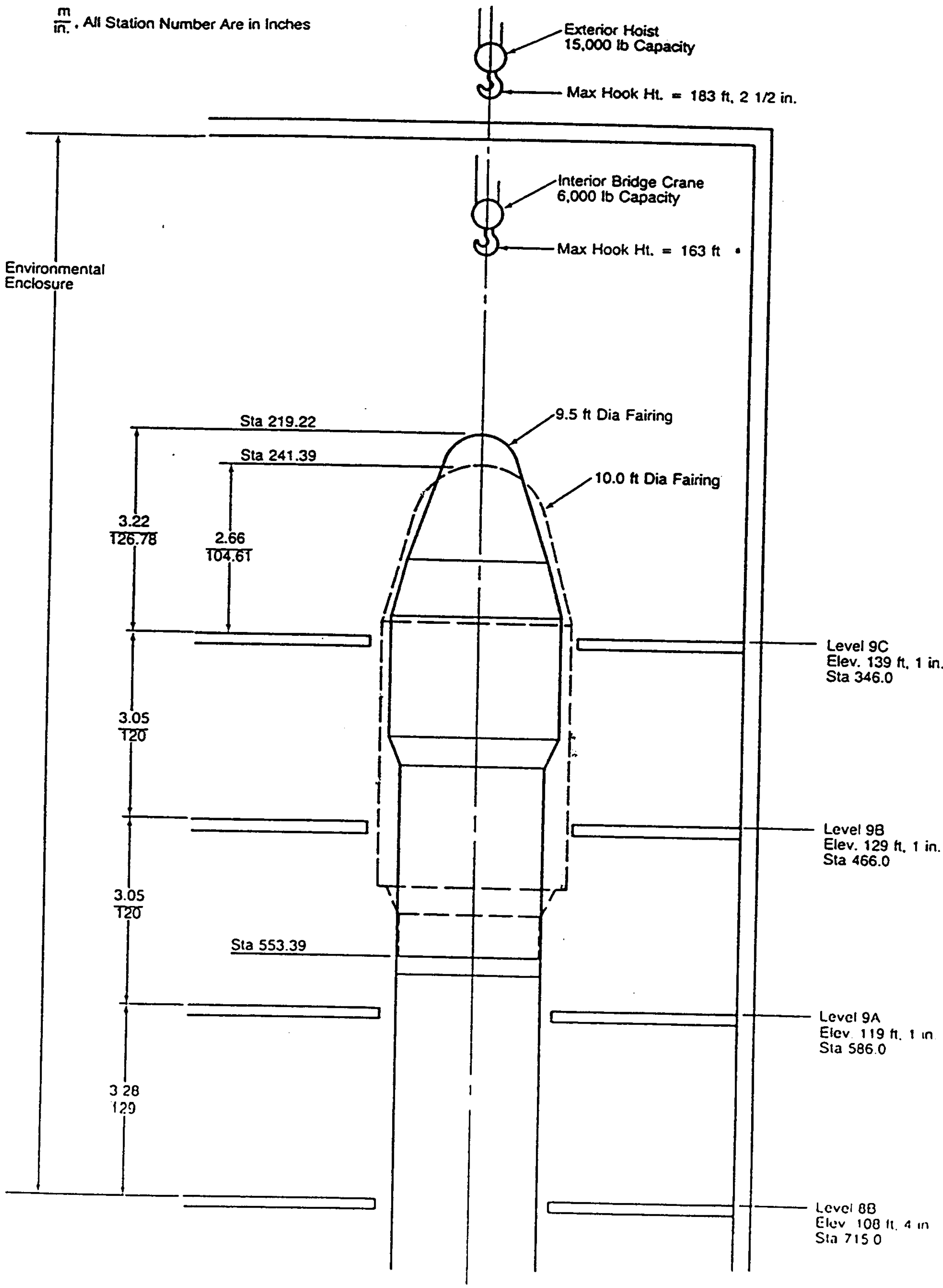
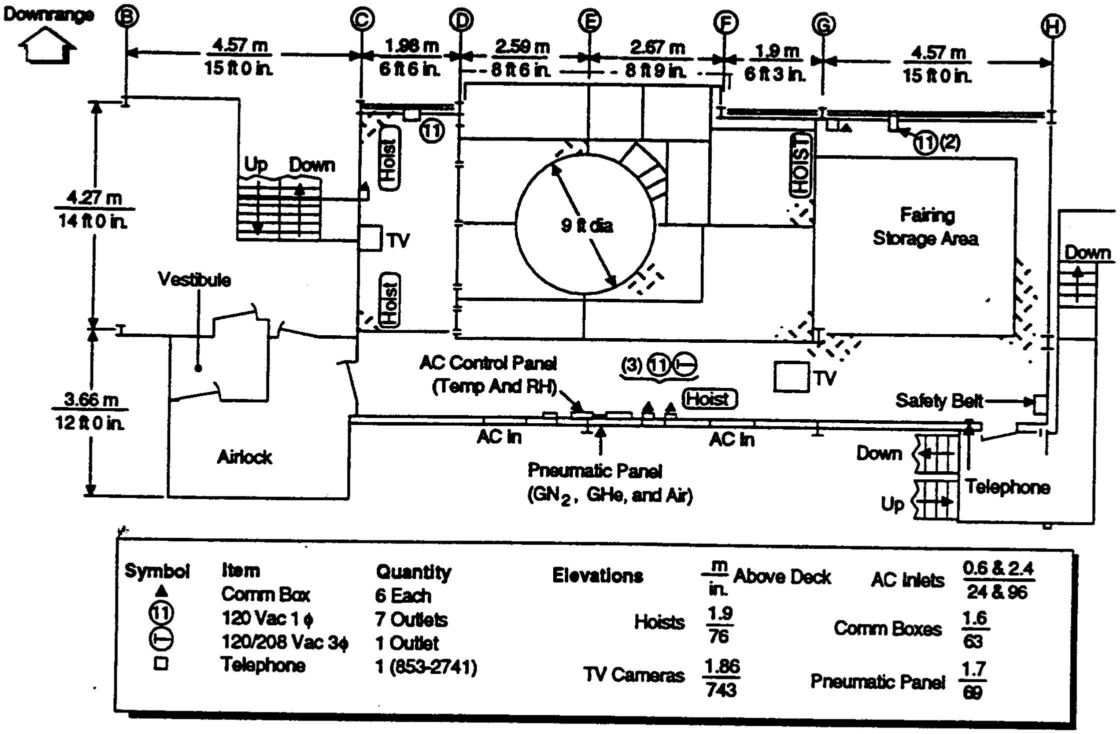


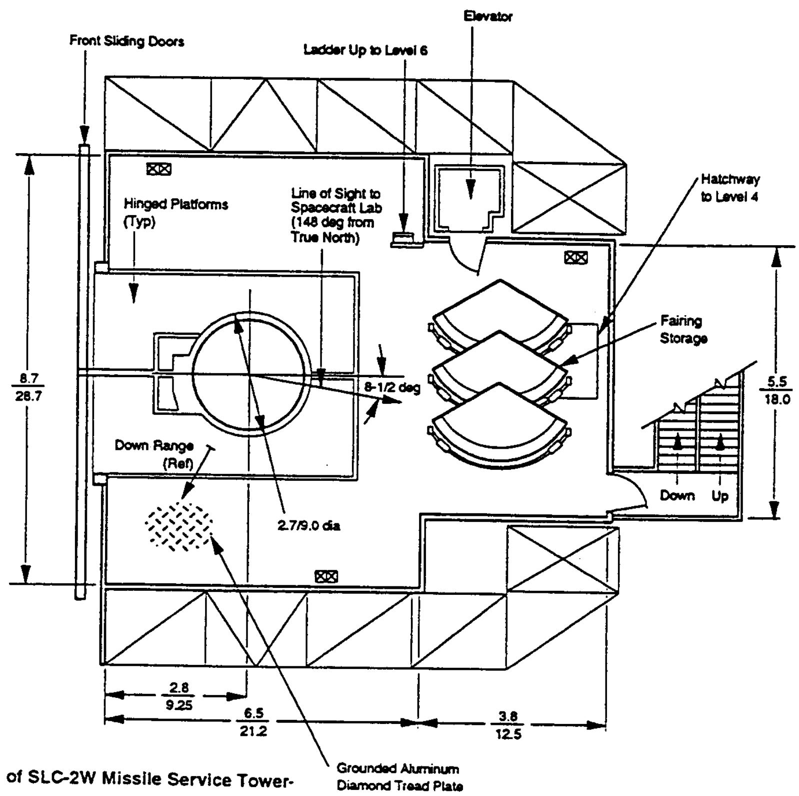
Figure 6.1h "Iron Maiden" Installation Fixture



Figurer 6.2.1a MST Environmental Enclosure Work Levels



Level 9A Floor Plan, Pads 17A and 17B



Level 5 of SLC-2W Missile Service Tower-

Figure 6.2.1b MST Floor Plans for CCAFS and VAFB

Section 7
Launch Operations

7.1 ORGANIZATIONS

A spacecraft - coordinator from the MDA-CCAFS launch team is assigned for each mission to arrange for support of the spacecraft, assist in obtaining safety approval of the spacecraft test procedures and operations, and integrate the spacecraft operations into the launch vehicle activities. KSC provides a Launch Site Support Manager. to assist the SP project during launch preparations.

7.2 DELTA II LAUNCH OPERATION SCHEDULES AND MEETINGS

During the launch scheduling preparation various meetings take place. Some of these meetings will require user input while others allow the user to monitor the progress of the overall mission. The MDA spacecraft coordinator will assure adequate user participation.

7.2.1 Countdown Schedule

The countdown schedule provides a detailed hour-by-hour, breakdown of the last several days. The countdown schedule illustrates the flow of activities from spacecraft erection through terminal countdown, and includes inputs from the spacecraft project. This schedule is the integrating document to ensure timely launch pad operations. A typical countdown sequence is shown in Section 6, Figure 6.1.

7.2.2 Delta Status Meetings

Status meetings are generally held twice a week. These meetings include a review of the activities scheduled and accomplished since the last meeting, a discussion of the problems and/or their solutions, and a general review of the mission schedule and specific mission schedules. Spacecraft representatives are encouraged to attend these meetings.

7.2.3 Spacecraft Schedules

The spacecraft project will supply schedules to the MDA spacecraft coordinator who will arrange support as required.

7.3 OPERATIONAL SAFETY

Safety requirements are covered in Section 9 of this manual. In addition, it is the operating policy that all personnel will be given safety orientation briefings prior to entrance to hazardous areas such as Complex **17.** These briefings will be scheduled by the MDA spacecraft coordinator and presented by the appropriate safety personnel.

7.4 SECURITY

7.4.1 Launch Complex Security

Complex 17 physical security is ensured by perimeter fencing, guards, and access badges. Security is provided on the MST by special access badges and a guard at the entrance to the spacecraft cleanroom. The access badges are distributed to those spacecraft and MDA personnel mutually agreed to by the spacecraft project and the MDA spacecraft coordinator. Typically, SP personnel are required to be escorted. Escorts are arranged by the LSSM.

7.4.2 CCAFS and VAFB Security

For access to CCA-FS or VAFB, US citizens or nationals must provide name, social security number, company name, and dates of arrival and departure to the MDA spacecraft coordinator and the Launch Site Support Manager. Arrangements will be made for entry authority. Non-US citizens must also provide nationality, date and place of birth, passport number with date and place of issue, visa number with date of expiration, and title or job description. Information for non-US citizens must be furnished at least two weeks prior to arrival.

7.5 COUNTDOWN AND LAUNCH TEAM ORGANIZATION

The Mission Director's Center (MDC) provides the necessary seating, data display, and communication to control the launch process. Figure 7.1 shows the mission control team organization for a NASA launch. Figure 7.2 shows a SP mission control team organization for a USAF launch. At the MDC the launch decision process is made by the appropriate management personnel representing the primary spacecraft, the launch vehicle, and the range. SP personnel are not located in the MDC for launch and are not a part of the official launch decision process. An input to the process for the SP is sometimes implemented. In this case, a launch decision process will be defined for the SP, and the input on SP readiness is given to the Delta Launch Director. The Launch Director has the authority to launch even if the SP is not ready.

The blockhouse provides the ground equipment, data display, and communication necessary to control, evaluate, and launch the Delta H.

7.6 PRELAUNCH REVIEW PROCESS

Periodic reviews are held to ensure that the launch vehicle is ready for launch. Figure 6.la shows the review relationship to the launch vehicle assembly and test cycle.

The following paragraphs discuss the Delta II readiness reviews. Participation of the SP organization is typically not required or is very limited.

7.6.1 Postproduction Review

This meeting, conducted at Pueblo, Colorado, reviews the flight hardware at the end of production and prior to shipment to CCAFS. No SP participation is required.

7.6.2 Pre-Vehicle-On-Stand (VOS) Review

This review is held at Huntington Beach approximately 3 months prior, to launch. This review includes a mission specific drawings, studies, and analyses as well as an update of the activities since Pueblo, the results of the DMCO processing, and any hardware history changes. KSC ELV Program Office normally provides a readiness statement for the SP.

7.6.3 Mission Readiness Review

This review is only for USAF primary missions. It is held prior to spacecraft erection at Los Angeles AFB. OLS and USAF SMC provide a brief description of the SP and a readiness statement.

7.6.4 Launch Site Readiness Review

This review is held prior to erection and mate of the upper stage and spacecraft. This review includes an update of the activities since the PreVOS review and verifies the readiness of the launch vehicle, third stage, launch facilities, and spacecraft for transfer of the spacecraft to the pad. The KSC launch operations manager typically provides a readiness statement for the SP.

Updated chart to go here

Figure 7.1 Mission Control Team Organization for NASA Primary Launches

Updated chart to go here

Figure 7.2 Mission Control Team Organization for SPs on USAF Launches

7.6.5 Flight Readiness Review

This review is conducted to determine that checkout has shown the launch vehicle and spacecraft are ready for countdown and launch. Upon completion of this meeting, an okay to load second-stage propellants is given. This review also statuses the range readiness to support launch and provides a predicted weather status.

7.6.6 Launch Readiness Review

This review is held on L-1 day and all agencies and contractors are required to provide a ready to launch statement. The SP agency typically provides a brief overview of the SP and its mission. Upon completion of this meeting, an okay to enter terminal countdown is given. The SP organization and KSC will provide a readiness statement which accounts for the SP. However, launch readiness may be approved even if the SP is not ready.

Section 8
Documentation

Effective integration of the spacecraft into the Delta launch system requires the diligent and timely preparation and submittal of certain documentation. When submitted, these documents represent the primary communication of requirements, safety data, system descriptions, etc., to each of the several support agencies. The payload project appoints a single individual to act as the interface with the KSC ELV Mission Integration Manager, who in turn acts as a single-point interface with the MDA integration manager for the mission. All data, formal and informal, is routed through these single-point interfaces.

Secondary Spacecraft documentation requirements are listed and described in Table 8.1 and must be available in a time consistent with the launch date. A typical schedule of documentation is shown in Figure 8.2. A specific schedule will be provided to the SP organization after MDA is given Authority to Proceed (ATP).

To facilitate the preparation of advance planning, certain documentation is required. The first step in the process is the Spacecraft Questionnaire. The spacecraft agency shall submit a response to the spacecraft questionnaire as soon as mission definition is reasonably firm, normally not later than 2 years before launch.

A definitive response to some questions may not be feasible since many items are defined at a later date. This is intended to be a one-time submittal to serve as a checklist for early mission planning.

Milestone	Agency	Description
1	SP	**SPACECRAFT QUESTIONNAIRE** • **Ini**tial definition of spacecraft description & requirements, interface Requirements and mission description. • First SP input; used as basis for First Mission Specif ication (Milestone 6).
2	SP	**SPACECRAFT DRAWINGS** • Should show nominal and worst-case dimensions from I/F plane of over-all size, protrusions, and hole patterns. • Preliminary drawings desired with S/C Questionnaire (M/S 1 above); others as early as possible. • Used for compatibility drawing (M/S 16), fairing clearance drawing (M/S 31) and various interface definition. • Final drawings required no later than 52 weeks before launch.
3	SP	**MASS PROPERTIES** • Mass properties are required to support various documents & analyses during the course of the mission integration, and should be supplied & updated in a timely manner (see schedule). • The various components of the SP should also be included in a weight summary, with an indication of the maturity of the weight value (i.e. estimated, calculated, weighed, etc.) • Summary to include total current, expected, and maximum weight; max and min MOI'S, POI's and C.G.'s
4	MDC	**KICKOFF INTEGRATION MEETING** • Shortly after receipt of the SP spacecraft questionnaire, Boeing will host a kickoff integration meeting. • The meeting serves as an introduction of the integration teams, of the integration process, the LJV, the SP and the primary & secondary missions.
5	OLS	**ATP** • **OLS provides** contract turn-on **of MDA for SP** integration.
6	MDC	**MISSION SPECIFICATION** • Contains all mission-peculiar requirements: s/c description, compatibility drawing, targeting criteria/orbit requirements, s/c requirements affecting the standard L/V, L/V description, special AGE & facilities. • Serves as Interface Control Document. • Initial issue published approximately 1 month after ATP; subsequent issues as necessary.
7	SP	**MISSION SPECIFICATION REVIEW AND CONCURRENCE** • S/C agencies review and concur within 30 days of completion of each issue; KSC/ELV provides signature once comments are incorporated. Formal issue then published. SP Agency does not sign the Mission Specification
8	SP	**S/C DYNAMIC MODEL** • SP contractor to provide a preliminary mathematical dynamic model to MDA. • MDA couples the bracketry model (item 9) to SP model for use in preliminary load factor evaluation and to assess the SP system dynamic characteristics. • Final SP dynamic model, usually including SP interface brackets, is correlated to modal survey test results (Item 12) and is delivered to MDA for use in coupled loads analysis.

Table 8.1 Secondary Payload Documentation Requirements

Milestone	Agency	Description
9	MDC	**INTERFACE BRACKETRY DYNAMIC MODEL** • A mathematical model of the SP interface bracketry structure joining the SP to the Delta guidance section is made early on in the program, including all MDA structure specific to the SP mission (e.g. longerons, fitting, clampband). • Bracketry model provided to SP agency for inclusion into the SP system dynamic model.
10	SP	**SIC MODAL SURVEY TEST PLAN** • SP agency submits a modal survey test plan for review by Mt)A and primary SIC contractor. • Test plan to include expected measurement locations, expected excitation locations and method (i.e. stinger, shaker table, calibrated impact hammer), and excitation levels as required. • It is recommended to include the MDA bracketry in modal survey test. • It is recommended to perform a fixed interface test, in lieu of a free interface test, to best simulate the boundary conditions of the SP system mounted to the Delta guidance section.
11	SP	**SYSTEM PRETEST MODEL** • SP system dynamic model, including any MDA bracketry involved in modal survey test and fixture flexibility, is used to identify measurement points, excitation points, and target modes for modal survey test. • Pretest model is provided to MDA and primary SIC agency for evaluation.
12	SP	**SIC MODAL SURVEY TEST REPORT** • Timely presentation of results is essential to allow initial approval prior to test specimen disassembly. • SP agency submits a modal survey test report for review by Boeing and primary S/C contractor. • Typical data in report includes: definition, sufficient transfer function plots to be able to identify modes (including excitation point), and linearity plots. • If not included in the modal survey test report, another report defining the SP dynamic model correlation to the modal survey test results is required.
13	MDC	**COUPLED DYNAMIC LOADS ANALYSIS** • Coupled dynamic analysis if performed in order to define maximum expected flight loads for Delta vehicle, primary SIC, and SP. • Coupled loads analysis events include: Ifftoff, transonic, max-q, and as required pre-MECO and MECO. • Output from analysis includes tables of maximum acceleration at selected nodes and/or internal loads of the SP and relative deflections (SP internal or with Delta fairing). As a minimum, the SP response recovery will include maximum SP cg load factors and relative deflections with the Delta fairing.
14	SP	**PRELIMINARY MISSION REQUIREMENTS** • Should be supplied as early as possible; at least 2 months before Preliminary Mission Assessment (M/S 15). • Will define desired orbit, tracking & T/M requirements, sequencing and pointing requirements. • Provides Indication of priority or degree of necessity. It may not be possible to meet all requirements.
15	MDA	**PRELIMINARY MISSION ASSESSMENT** • Confirm capability of primary mission orbit/vehicle performance to perform the secondary mission requirements. • Evaluate orbit requirements, tracking requirements, sequencing. • Assessment phased with Primary Mission Preliminary Mission Analysis (PMA); approximately - 1 year before launch.

Table 8.1 Secondary Payload Documentation Requirements (continued)

Milestone	Agency	Description
16	**MDC**	**S/C COMPATIBILITY DRAWING** • Combines SP and launch vehicle drawings to show all pertinent mechanical interfaces.
17	**MDC**	**S/C SEPARATION ANALYSIS** • Performed to verify adequate clearance between second stage and SP during separation. • Requires full set of mass properties defined in M/S 3. Lateral cg offset and spring energy uncertainty have the greatest affect on angular tipoff rate. • Provides clearances, angular tipoff rates, and separation velocity. • Preliminary analysis performed, if necessary, to address issues concerning clearance, angular tipoff rate and separation velocity.
18	**SP**	**RADIO FREQUENCYAPPLICATION** • The SP agency is required to specify the RF transmitted by the spacecraft during ground processing and launch intervals. • MDA will integrate the SP and L/V RF data into an RF application for the mission and submit to the appropriate range/government agencies at least 1 year before launch.
19	**MDC**	**MODIFICATION HARDWARE DESIGN REVIEW** • Review all mission-specific vehicle mods at least 1 month prior to implementation. • Review all preliminary analysis to date. • USAF/KSC/ELV approval required to proceed with vehicle mods.
20	**SP**	**S/C ENVIRONMENTAL TEST PLAN AND TEST REPORT** • Test plan defines approach for secondary payload qualification and acceptance testing. • Includes test philosophy, test objectives, general test methods, and test schedule. • Specimen must be in final flight configuration or a list of deviations must be submitted for approval. • Test report documents results of testing and analyses that demonstrate adequacy of secondary payload for flight environments and loads. A test data summary sheet must be submitted within two weeks of test completion.
21	**SP**	**S/C PROGRAM REQUIREMENTS DOCUMENT** • Defines the SP range and network support requirements for pre-flight testing and on orbit operations. • Support is coordinated by NASA/GSFC Code 500. • Standard UDS format is used. • Complex mission requirements may need coordination earlier than 1 year before launch.
22	**MDC**	**DELTA PRD/OR** • **Defines range and network requirements for the L/V from liftoff thru SV separation and depletion burn. Furnished to ER 6 months before launch**
23	SP	S/C MATERIALS LIST • SP program must submit a complete list of materials used in the payload; copies of drawing parts lists are acceptable. • Materials to be selected in accordance with NASA specifications JSC SP-R-0022A on outgassing contamination, and within standards of practice for space hardware selection. • The materials list will be transmitted with a cover letter which will note any materials which violate the above requirements.
24	**MDC**	**SP TO VEHICLE FITCHECK** • A fitcheck with the SP flight hardware interfaces and the flight vehicle interfaces will be conducted at MDA/ Pueblo during the L/V production process. • Purpose is to confirm fit/clearance of the flight hardware, and ability to install SP hardware at the launch site. Schedule is driven by flight vehicle production schedule.

Table 8.1 Secondary Payload Documentation Requirements (continued)

Milestone	Agency	Description
25	**SP**	**S/C MSPSP (ARAR)** • Data package: detail all hazardous and safety critical systems/subsystems and their interfaces in vehicles, payloads, ground support equipment, facilities and launch pads. • Provides verification of compliance with ERR 127-1 or WRR 127-1 requirements. • Must be approved by 45SPW or 30SPW safety organization prior to the arrival of any payload element, activation of a hazardous processing facility or commencement of any hazardous operation on ER or WR.
26	**SP**	**S/C LAUNCH SITE TEST PLAN** • **A** description of all activities and operations conducted while the SP is at the launch site. • It provides all agencies with a detailed understanding of these activities, and is submitted to and approved by the NASA Launch Site Support Manager (LSSM). • A draft of this plan must be submitted approximately 1 month prior to the first Ground Operations meeting at CCAFS (M/S 27).
27	**MDC**	**GROUND OPERATIONS MEETING** • This meeting (at the launch site) provides a forum for discussion of SP and L/V activities at the launch site. • Logistics, resource requirements, and integrated (joint SP-L/V) testing requirements such as Delta Mission Checkout testing, F12 (simulated flight test) and F6 (flight program verification and power-on voltage) testing are discussed.
28	SP	**LAUNCH VEHICLE INSIGNIA** • The SP project is entitled to place a mission insignia on the L/V. • The proposed design must be submitted no later than 9 months before flight. • Exact graphic design and color definition is required (8'xlO"color photo graphs as a minimum). • The maximum size on the L/V will be 4'x4'(and may be smaller).
29	SP	**DETAILED TEST OBJECTIVE REQUIREMENTS** • Final input to NASA of SP Detailed Test Objectives (DTO) orbit/mission requirements. • Defines (for example) desired mission orbit and allowable variations, special pointing or attitude requirements, sequencing, etc. • Supports development of the DTO extension (M/S 30). • Provided no later than 9 months before launch.
30	MDC	**DTO EXTENSION** • Beginning after the Primary Payload deployment (the end of the primary mission DTO), this report "extends' the Primary Payload DTO for the SP mission description. • Contains SP mission description, nominal trajectory printout, sequence of events, tracking/telemetry coverage summary and other pertinent information. • Available approximately 6 months before launch.
31	MDC	**FAIRING CLEARANCE STUDY DRAWING** • Confirms acceptable clearance between SP and fairing with consideration of relative deflections and tolerances. • Final definition of SP size/configuration is required. • Envelope exceedances may be evaluated by measurement at CCAFS. • Unacceptable clearances would preclude flight on the designated mission.
32	MDC	**FINAL ANAYSIS TIM** • MDA will host a Technical Interchange meeting (TIM) to review all SP and L/V final analysis and test results. • This review supports the final USAF/OLS risk assessment, and takes place 3-4 months before launch.

Table 8.1 Secondary Payload Documentation Requirements (continued)

Milestone	Agency	Description
33	**SP**	**S/C STRENGTH REPORT** • Purpose is to insure that the SP has no detrimental effect on the primary mission. • Assesses the structural integrity of customer-furnished payload components under worst case loading environments. A venting analysis must be included. • Provides margin of safety calculations and identifies areas of secondary payload hardware where minimum margins occur.
34	**MDC**	**VEHICLE STRENGTH REPORT** • **Assesses structural integrity of all launch vehicle components for combined primary and secondary payload loading environments.** • Provides margin of safety calculations and identifies critical areas of launch vehicle structure where minimum margins occur.
35	**SP**	**S/C LAUNCH SITE TEST PROCEDURES** • **SP operating procedures must be prepared for all launch site operations.** • Special instructions for preparation of hazardous procedures (M/S 25) must be followed. • Procedures must be received at least 75 days prior to use.
36	**SP**	**S/C INTEGRATED TEST PROCEDURE INPUTS** • **MDA prepares launch site procedures for operations involving the SP and its interrelationship with the Delta upper stage.** • SP inputs with regard to special handling, transportation, installation/mating with the L/V, and testing such as DMCO, F12, F6 and final closeout are required.
37	**MDC**	**INTEGRATED COUNTDOWN SCHEDULE** • **Description of daily, hourly flow at LC17 for L/V activities from S/C erection to launch.** • SP activities are included (based on SP inputs to MDA), and time will be allocated in these schedules.
38	**MDC**	**DMCO AND GROUND OPERATIONS MEETING** • **The SP electrical system interfaces, if any, will be included in Delta Mission Checkout (DMCO) during the Dual Composite flight simulation at CCAFS.** • This testing typically occurs 3-5 months before flight. • This opportunity is also taken to conduct a final Ground Operations Meeting (see M/S 27) prior to SP arrival at LC17.
39	**MDC**	**LAUNCH SITE PROCEDURES** • **MDA creates a launch site procedures document for L/V pre-launch checkout.** • MDA will accommodate SP requirements defined in previously submitted S/C procedures (see M/S 35 & 36).

Table 8.1 Secondary Payload Documentation Requirements (continued)

INTEGRATION AND DOCUMENTATION SCHEDULE

Agency	Item No.	Milestones	MONTHS PRIOR TO LAUNCH: 24 23 22 21 20 19	18 17 16 15 14 13	12 11 10 9 8 7	6 5 4 3 2 1 L
SP		Spacecraft Questionnaire	Δ			
SP		Spacecraft Drawings	Δ		Δ	
SP		Mass Properties Statement	Δ	Δ	Δ	Δ
MDC		Kick Off Integration Meeting	Δ			
OLS		ATP		Δ		
MDC		Mission Specification		Δ (update	as required)	
SP		Mission Spec Review/Concur		Δ		
SP		S/CDynamic Model		Δ	Δ (test	verified)
MDC		I/F Bracketry Dynamic Model		Δ		
SP		S/C Modal Survey Test Plan		Δ (1.5	months prior	to test)
MDC		System Pretest Model		Δ		
SP		S/C Modal Survey Test Report			Δ	
MDC		Coupled Dyn Load Analysis				Δ
SP		Preliminary Mission Reqmts		Δ Δ		
MDC		Preliminary Mission Assessment			Δ	
MDC		S/C Compatibility Drawing		Δ		
MDC		S/C Separation Analysis		Δ		
SP		RFA			Δ	

Figure 8.2 Secondary Spacecraft Documentation Requirements (Continued)

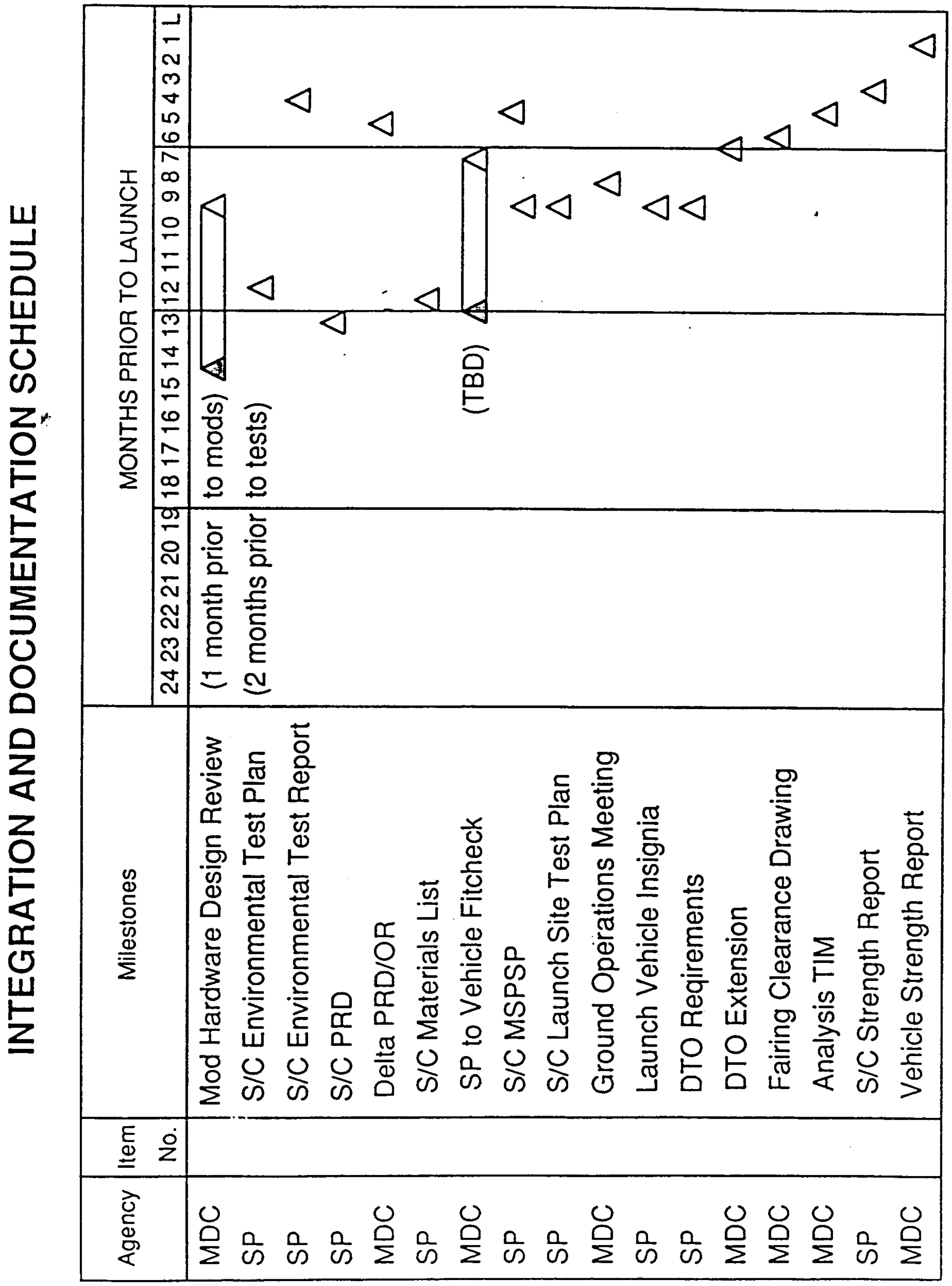


Figure 8.2 Secondary Spacecraft Documentation Requirements (Continued)

INTEGRATION AND DOCUMENTATION SCHEDULE

Agency	Item No.	Milestones	MONTHS PRIOR TO LAUNCH 24 23 22 21 20 19	18 17 16 15 14 13	12 11 10 9 8 7	6	5	4	3	2	1	L
SP		S/C Launch Site Test Procedures					△					
SP		S/C Integrated Test Procedure Inputs					△					
MDC		Integrated Countdown Schedule						△				△
MDC		DMCO and Ground Operations Meeting						△				
MDC		Launch Site Procedures								△		

Figure 8.2 Secondary Spacecraft Documentation Requirements (Continued)

Section 9
Safety

Delta II pre-launch operations for NASA spacecraft are conducted at CCAFS ,and Kennedy Space Center, Florida, and at VAFB, California, by arrangement with the appropriate agencies. The USAF is responsible for the overall safety at CCAFS and VAFB and has established safety requirements accordingly. Before a spacecraft moves onto USAF property, the spacecraft agency must provide the range safety office with certification that its system has been designed and tested in accordance with range safety requirements (ESMCR or WSMCR 127-1, Chapters 3 and 5). Safety of operations conducted at NASA facilities on KSC must comply with KSC Management Instruction (KMI) 1710.lE; for operations on CCAFS, both range safety and NASA safety requirements are applicable. The NASA safety requirements and quality assurance (SR&QA) and protective services directorate implement the KSC safety program for operations conducted at NASA facilities. In general, USAF and NASA safety regulations for pre-launch activates are equivalent. Since many spacecraft operations are performed by or involve launch vehicle contractor personnel, certain additional NIDA safety requirements also apply.

The following documents specify the safety requirements applicable to Delta H users:

A. ESMCR 127-1, Range Safety Manual, with revisions issued to date (ER users only, Reference 2).

B. WSMCR 127-1, Range Safety Manual, with revisions issued to date (V%TR users only, Reference 3).

C. KMI 1710.lE, Safety, Reliability, and Quality Assurance Programs, dated 11 April 1985.

9.1 DOCUMENTATION REQUIREMENTS

Both USAF and NASA require formal submittal of safety documentation containing detailed information on all hazardous systems and associated operations. ER requires preparation and submittal of a Missile System Pre-launch Safety Package (MSPSP). The formal requirements of this document are found in ESMCR 127-1 (Reference 2). WR requires preparation and submittal of an Accident Risk Assessment Report (ARAR); see Reference 3. Generally, data requirements include design, test, and operational considerations. NASA requirements in almost every instance are covered by the USAF requirements; however, the spacecraft agency can refer to KMI1710.lE for details.

Each of the USAF support organizations at ER and WR retain final approval authority over all hazardous operations that take place within the confines of their respective jurisdictions. Therefore, the spacecraft agency should consider the requirements of the 127-1 manuals and KMI 1710.lE from the outset of a program and should submit the required data at the earliest possible date.

The MSPSP is submitted to the appropriate government agency for review and further distribution. Twelve copies of the original and all revisions are required. The review process usually requires several iterations until the system design and its intended use are considered to be in compliance with all safety requirements. The flow of spacecraft safety information is shown in Figure 9.1 for Eastern Range operations.

9.2 HAZARDOUS SYSTEMS AND OPERATIONS

The requirements cited in 127-1 apply, but are augmented as follows for design and operations requirements. MDA safety requirements do not permit MDA personnel to be exposed to leak, functional, or operational testing of pressure vessels or systems at a burst-to-operating pressure ratio of less than 4 to 1, except on the launch complex and in the spin facility where a safety factor of 2 to 1 is acceptable for vessels. Therefore, for spacecraft operations (except in the noted facilities) where MDA personnel are required to be present, the 4 to 1 safety factor applies. Spacecraft that contain pressure vessels or systems with a safety factor less than 4 to 1, should plan on final pressurization of the vessels or systems on the launch complex by the complex high pressure gas system. The design and operations requirements governing activities involving MDA participation are discussed in the following paragraphs.

9.2.1 Functional or Leak Tests in HPFs on Systems with 4 to 1 Safety Factors

Vessels designed with a minimum calculated burst pressure of four times maximum allowable operating pressure which also meets the requirements of 127-1 and have been proof tested to at least one and one-half times maximum operating pressure are acceptable.

9.2.2 Functional or Leak Tests in Shop Areas on Systems with Less Than 4 to 1 Safety Factors

For vessels with a minimum calculated burst pressure of less than four times maximum calculated allowable operating pressures, but which meet the requirements of 127-1 or where the test pressure shall not exceed one-fourth of the minimum calculated design burst pressure, are acceptable. A proof test shall have been performed on each vessel to at least one and one -half times the test pressure.

SPACECRAFT SAFETY DOCUMENTATION
(EASTERN RANGE)

Notes

1. KSA Safety approval is required for operations in NASA facilities prior to Range Safety /SE review.
2. Space Launch Squadron (SLS) and launch service contractor review is for hazardous documentation which affects USAF and contractor facilities only.
3. Coordinates comments and status.
4. Includes all spacecraft documentation such as MSPSP, LSTP, Spacecraft Hazardous Procedures, etc.
5. ESMCR 127.1 is the governing safety document for all CCAFS operations.

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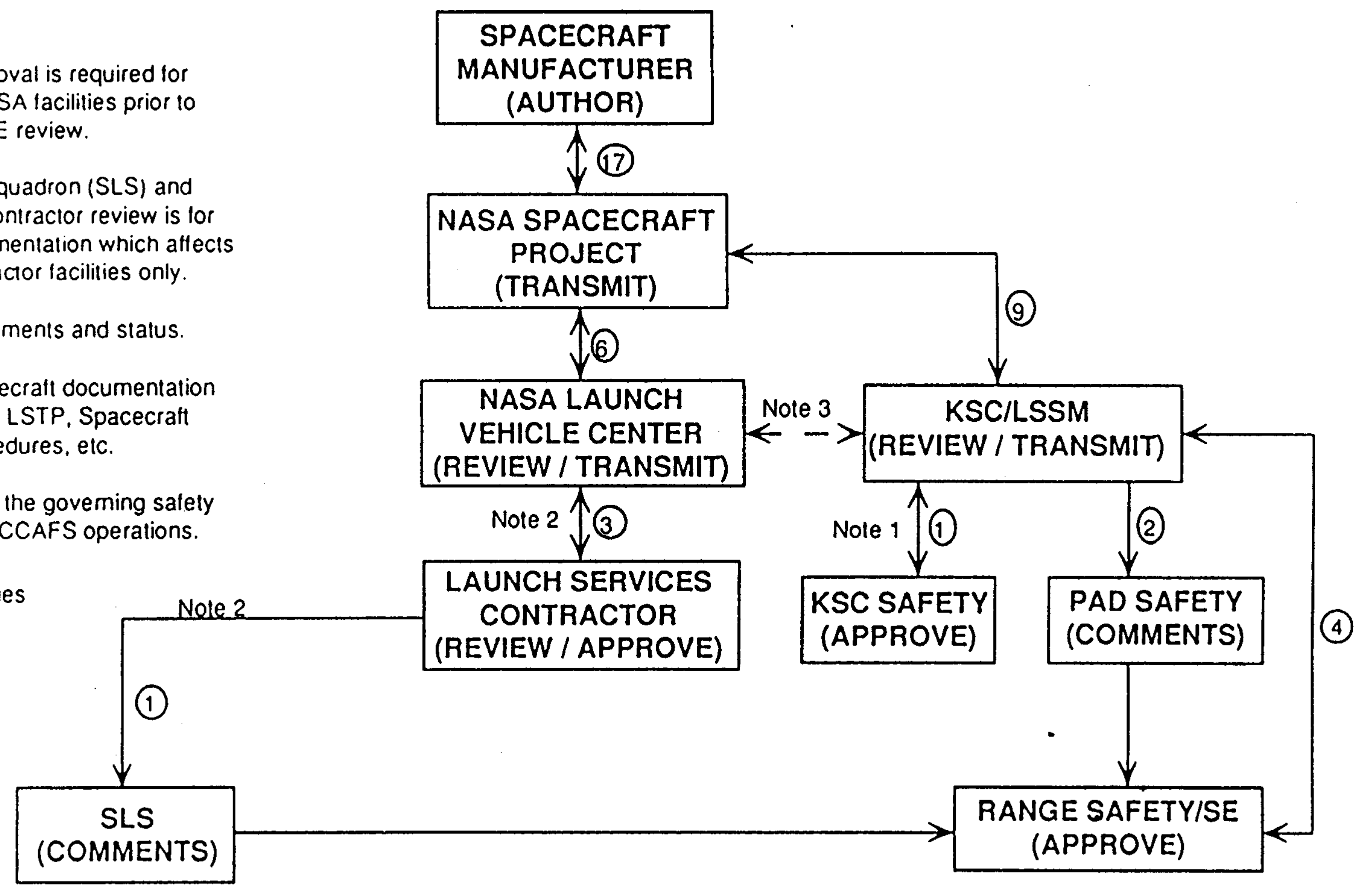

Figure 9.1 Spacecraft Safety Documentation

9.2.3 Operations in the HPFS, Transportation to the Launch Complex, and Operations on Systems with Less Than 4:1 Safety Factors.

Hazardous pressure vessels are those which have a burst-before-leak failure mode or which contain hazardous fluids. In order for MDA personnel to be exposed to hazardous pressurized vessels, the vessels must be designed, built and tested to meet minimum factor of safety requirements (ratio between maximum operating pressure (MOP) and design burst pressure). All-metal vessels designed with a minimum calculated burst pressure of four times maximum allowable operating pressure which also meet the requirements of TSMCR 127-1 are acceptable. Metal tanks designed, built and verified according to MIL-STD-1522A, Approach A, with a minimum safety factor of 2:1 are also acceptable for MDA personnel exposure. Personnel exposure to non hazardous leak-before-burst metallic vessels may be permitted with safety factors as low as 1.5:1. Any operation which requires pressurization above levels which would maintain the 4:1 safety factor must be conducted remotely (no personnel exposure) and requires a minimum five minute stabilization period prior to personnel exposure.

Increasing use of composite pressure vessels in spacecraft propellant systems has led to the adoption of revised safety factor requirements for these types of pressure vessels. Safety factors applicable to composite pressure vessels differ from those applicable to all-metal tanks. The required safety factor varies according to the design and construction of the tanks. The applicable safety factor for vessels of composite construction is dependent upon the percentage of load carried by the overwrap. A joint policy letter was issued by the launch ranges containing safety factor requirements for tanks of various design and construction. MDA requirements have been revised to be aligned with the launch range requirements. Analyses and test documentation which verifies the pressure vessel safety factor must be included in the spacecraft safety documentation.

9.2.4 Nonionizing Radiation

The spacecraft nonionizing radiation systems are subject to the design criteria in the USAF and KSC manuals and the special Delta-imposed criteria as follows: systems producing nonionizing radiation will be designed so that the hazards present to personnel are at the lowest practical level. McDonnell Douglas employees are not to be exposed to nonionizing radiation above 1 mW/cm2 averaged over any 1-min interval.

9.3 WAIVERS

Range safety organizations discourage the use of waivers to safety requirements. They are normally granted only to spacecraft designs which have a proven safe history. After a complete review of all safety requirements, the spacecraft agency must determine if waivers are necessary. A waiver is required to any safety-related requirements that cannot be met. Requests for waivers shall be submitted prior to implementation of the safety-related requirements in question. Waiver requests must be accompanied by sufficient substantiating data to warrant consideration and approval. It should be noted that the USAF has final approval of all waiver requests and that no guarantees are made that approval will be granted.

References

1. Commercial Delta II Payload Planners Guide, McDonnell Douglas Document MDC H3224B, December 1989.

2. Eastern Space and Missile Center Range Safety Requirements, EWR 127-1,.

Acronyms

ARAR	Accident Risk Assessment Report
CCAFS	Cape Canaveral Air Force Station
CDR	Command Destruct Receiver
EED	Electroexplosive Device
ELV	Expendable Launch Vehicle
ER	Eastern Range
DMCO	Delta Mission Checkout
DRIMS	Delta Redundant Inertial Measurement System
FUT	Fixed Umbilical Tower
GC	Guidance Computer
GSFC	Goddard Space Flight Center
GTO	Geosynchronous Transfer Orbit
HPF	Hazardous Processing Facility
IRIG	Inter-Range Instrumentation Group
KSC	Kennedy Space Center
LC	Launch Complex
MDC	Mission Director's Center
MDA	McDonnell Douglas Space Systems Company
MECO	Main Engine Cutoff
MSPSP	Missile System Prelaunch Safety Package
MST	Missile Service Tower
NCS	Nutation Control System
OASPL	Overall Sound Pressure Level
OLS	Orbital Launch Services
PAA	Payload Adapter Assembly
PAF	Payload Attach Fitting
PRD	Program Requirements Document
PSM	Program Support Manager
RF	Radio Frequency
RFI	Radio Frequency Interface
RIFCA	Redundant Inertial Flight Control Assembly
SECO	Second Engine Cutoff
SIRD	Support Instrumentation Requirements Document
SLC	Space Launch Complex
SP	Secondary Payload
SPE	Secondary Payload Envelope
SPI	Secondary Payload Interface
SRS	Shock Response Spectrum
TC	Test Conductor
TM	Telemetry
USAF	United States Air Force

VAFB	Vandenberg Air Force base
VOS	Vehicle on Stand
WR	Western Range